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AD NUMBER
AD083625
CLASSIFICATION CHANGES
TO: unclassified
FROM: confidential
LIMITATION CHANGES
TO: Approved for public release, distribution unlimited
FROM: Distribution authorized to U.S. Gov't. agencies and their contractors; Administrative/Operational Use; 03 NOV 1955. Other requests shall be referred to Office of Naval Research, Arlington, VA 22203.
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30 Nov 1967, DoDD 5200.10; ONR ltr, 28 Jul 1977

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MILITARY HELICOPTER TRANSPORT SYSTEMS 1956 to 1961

FOR THE
UNITED STATES ARMY
THROUGH THE
OFFICE OF NAVAL RESEARCH - AIR BRANCH

Summary Report No. 350.1
30 November 1955

ENGINEERING DIVISION HILLER HELICOPTERS

ERRATA

Page e—Summary—On twelfth line down from top of left hand column—change "five to ten" to "ten to fifteen."

Page 9—On line 13 of second paragraph of right hand column—change "Engineer Battalion" to "Engineer Company."

Page 18—On last line of second paragraph of left hand column—change "C-1 and C-2" to "C-1 through C-4."

Page 48—On eighth line down from top of left hand column—change "the five" to "three."

Page 50—Third line from bottom of left hand column—change "five" to "four."

Page 63—1. In Item F for Figure IX-3—change "H-16A" to "H-16"
2. In second line of third paragraph in right hand column—change "H-16A" to "H-16".

Page 64—In twelfth line down from top of left hand column—change "H-16A" to "H-16".

Page 74—Third line down from top of right hand column—change "consideraing" to "considering."

Page 88—In Figure D-3—change "take of" to "take off."

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MILITARY HELICOPTER TRANSPORT SYSTEMS 1956 to 1961

Summary Report No. 350.1
Contract No. Nonr-1340(00)

PREPARED BY
TRANSPORT HELICOPTER STUDY GROUP

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ENGINEERING DIVISION HILLER HELICOPTERS

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SUMMARY

THE PROBLEM

This report is the result of a broad parametric study to develop coherent technical, economic, and operational criteria for future Army helicopter transport systems. This concept evolved from an earlier interpretation of Contract Nonr 1340(00) which specified a design study of a three ton payload transport helicopter. It was considered that the parametric analysis approach might reveal significant criteria for *design optimization*, and should therefore precede specific design studies.

Objectives

The objectives are:

1) To establish *quantitative specifications* to serve as a guide for the time period 1956 to 1961, using helicopter types which could become production realities within this five year period.

2) To establish *projected indications* of the possibilities of helicopter types with power applied at the rotor tips, for which additional research and development programs could result in production within the time period 1960 to 1970.

3) To develop generalized *design analysis methods* required to fulfill objectives (1) and (2), and to synthesize these methods under separate cover in a report entitled *Transport Helicopter Design Analysis Methods*, for use by military procurement personnel in preliminary design analysis.

Assumptions

The technical, economic, and operational analyses embodied in the study are each predicated upon certain assumptions which have been subjected to a preliminary test of validity at a contract "Shredding Session" which took place at the Office of Naval Research in Washington, D.C., on June 28 and 29, 1955. The more important of these assumptions are summarized below.

A. *Technical:*

1) *Conventional helicopter design practice*, as exemplified by current production and prototype helicopters, is assumed as a general foundation, modified and augmented by anticipated state of the art improvements only in cases which are non-controversial, and based on thorough analysis rather than intuition.

2) *Generalized powerplant characteristics* are assumed for each powerplant type considered, based on state of the art statistics for existing operational types, and conservative estimates for advanced types in the development stage.

B. *Economic:*

1) *A fixed size of the Army* is assumed, with the result that costs for base establishment, supply, and military training are eliminated from the study.

SUMMARY

2) *Pipeline costs* for shipment of helicopter fleets and spare parts from the Zone of the Interior to a theatre of operations are omitted from the study.
3) *Helicopter combat attrition* is not considered in this study, but losses due to *operational attrition* are accounted for.

C. *Operational:*
1) *Full gross weight operation* is assumed for both the outbound and the inbound trips.
2) *A fixed aircraft availability* is assumed, hence flight utilization is varied with design payload in accordance with cargo loading and unloading time.
3) *Detailed hourly flight scheduling* is not considered, since the basic transport requirement is assumed to be the daily support of military combat elements.

Scope

The general vehicle type is restricted to the *pure helicopter*, relying entirely upon *conventional rotors* for lift and propulsion in all flight regimes, and arranged either in the *single rotor plus tail rotor* configuration, or in the *tandem* configuration.

Power plant types are restricted to *supercharged aircooled reciprocating* engines and *geared turbine* engines for the 1956-1961 time period, and to *tip-mounted ramjets, turbojets,* and *pressure jets* for the projected 1960-1970 time period. Applications of tip power are restricted to the single rotor helicopters.

Design payloads from one to five tons, design radii of action from 25 to 150 nautical miles, and design hover ceilings (in standard NACA atmosphere, out of ground effect, using normal rated power) from 5000 to 10000 feet are included in the scope of the basic study.

Within the framework of the 1956-1961 matrix are included supplementary special studies of the effects of retractable landing gear, externally carried payload, and hovering at extreme operating temperatures at 6000 feet altitude.

The Measure of Effectiveness

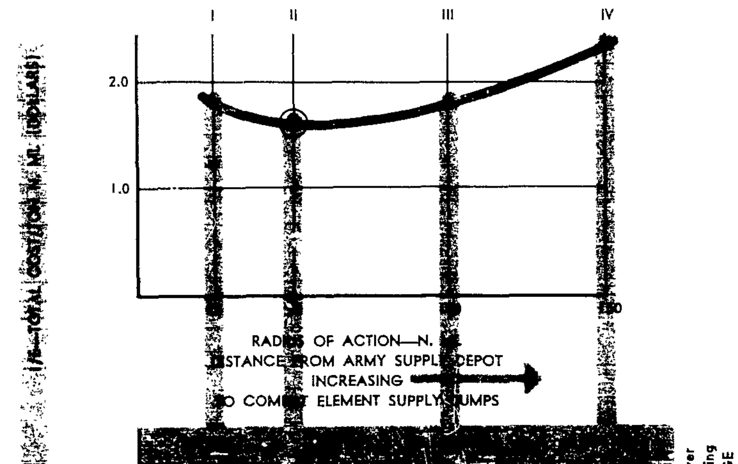
The military commander in a logistic supply situation is most concerned with *transport work capacity*, defined as ton-miles per hour. However, from the standpoint of the military planner, faced with the economic and budgetary problems of maintaining an adequate defense, the *cost* of this transport work capacity is of equal importance. *The Measure of Effectiveness E*, which forms the comparison criterion by which optimum helicopter transport systems are selected in this study, combines the two equally important factors of cost and work capacity, and is defined as the ratio of ton-nautical miles per flight hour to the total cost per flight hour. This measure is inverted in the graphical presentation of the results, taking the form 1/E, with dimensions reduced to *dollars per ton-nautical mile.*

RESULTS

Predominant Factors

The major determining factor which delineates the optimum helicopter transport system is the *misson radius of action.* The results of the parametric evaluation, without exception, illustrate that the selection of a design radius of action establishes, for maximum effectiveness, the remaining system parameters by virtue of the inherent technical, economic,

SUMMARY



Configuration	Single	Tandem	Single	Tandem	Single	Tandem	Single	Tandem	Hover Ceiling OGE
Payload (Tons)	3.2	4.3	3.8	4.5	4.2	4.5	3.8	4.3	5000 Ft.
Gross Weight (Lbs.)	18500	22700	23700	26500	32000	32700	37700	39500	
Installed Horsepower	2300	2950	2840	3240	3350	3610	3810	4120	
Payload (Tons)	3.3	4.2	3.6	4.4	3.7	4.4	3.6	4.0	7500 Ft.
Gross Weight (Lbs.)	19500	23200	23300	27200	30000	33800	38000	40000	
Installed Horsepower	285	3140	3040	3430	3570	3870	4350	4380	
Payload (Tons)	3.3	4.0	3.6	4.3	3.7	4.2	3.6	3.8	10000 Ft.
Gross Weight (Lbs.)	20000	23200	24000	27200	32600	34200	42000	42000	
Installed Horsepower	3170	3300	3310	3660	4020	4320	5030	5080	

NOTE: All systems are geared gas turbine powered

FIGURE ONE

and operational restraints imposed by the system.

A factor of lesser influence on the optimum system is the *design hover ceiling.* It is apparent from the results of the analysis that variation in design hover ceiling between 5000 and 10000 feet, standard altitude, causes no significant variation in the effectiveness of the optimum system. Extremes in hover performance from marginal sea level hover performance to hover ceilings greater than 10000 feet both result in rapidly increasing cost per ton nautical mile.

Decision Analysis 1956-1961

Considering the effect of variation in probable mission radii of action, the factor of primary influence, the decision analysis, Figure One, is presented for a range of values. Included within the scatter band, indicating the margin of probable error, are the optimum helicopter transport systems comprising both single and tandem rotor configurations for design hover ceilings between 5000 and 10000 feet. The tabulated values emphasize the relative insensitivity of the measure of effectiveness to a decision between single and tandem

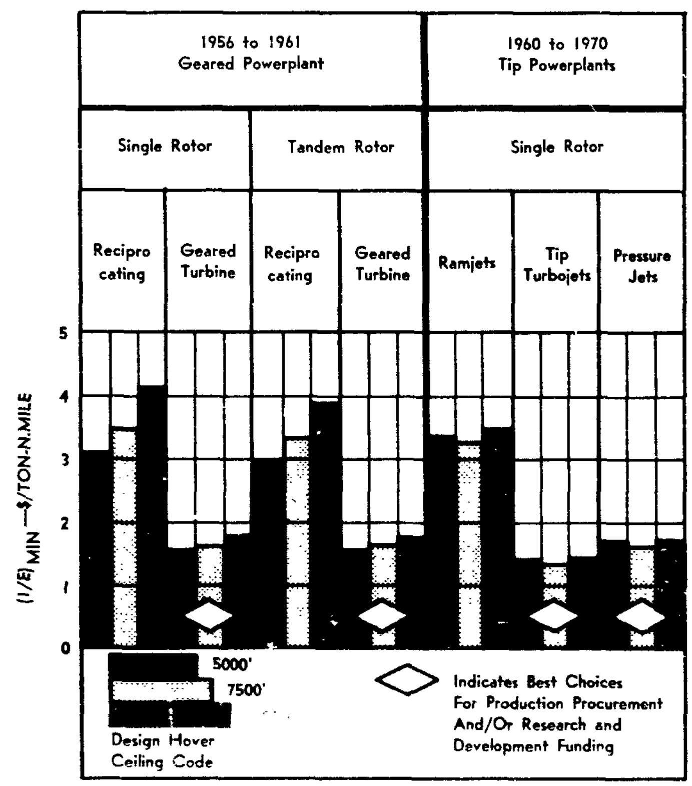
SUMMARY

rotors or to a selection of design hover ceiling in the range considered. For all systems included in the decision analysis, geared turbine power plants are clearly mandatory for lowest cost per ton-nautical mile. Only the combined effects of high design hover ceiling, temperature and radius of action produce a situation in which reciprocating engine powered helicopters become competitive.

For a selected radius of action, the optimum tandem rotor system displays a higher payload and correspondingly higher design gross weight than the optimum single rotor system. The higher payload increases work capacity, however, the higher gross weight increases costs proportionately such that negligible difference in cost per ton-nautical mile exists between the two rotor types.

Decision Analysis 1960-1970

Decisions for the advanced time period, 1960 to 1970, represent optimum avenues for research and development. This group of tip powered configurations is compared in Figure Two with the configurations of the 1956 to



COMPARISON OF GEARED-ENGINE HELICOPTERS (1956 TO 1961) AND TIP-POWERED HELICOPTERS (1960 TO 1970)

FIGURE TWO

SUMMARY

1961 period. Each value of cost per ton-nautical mile shown represents the cost at optimum design payload and radius of action for the configuration.

Tip turbojet and pressure jet systems appear as the best indications of future research and development funding, with a small superiority shown by the tip mounted turbojet. The ramjet powered helicopter does not appear to be competitive for the transport mission. Optimum radii of action for this configuration are of the order of five to ten nautical miles, and at these radii the cost per ton nautical mile is prohibitively greater than for tip turbojet or pressure jet systems.

Penalty Analysis

The choice of radius of action, the determining factor in selecting an optimum helicopter system, requires that the military planner weigh the relative import of many interrelated factors. Since these factors change with time, the anticipated mission radii will change, altering the selection of the helicopter transport system. Having selected a mission radius on the basis of projected military requirements, it is conceivable that the actual military environment at some later date may be altered to require a different helicopter system. Figure Three shows the penalty in transport effectiveness incurred by selection of a mission radius of action subsequently shown to be incorrect, and in addition, shows in what manner this penalty may be minimized.

Systems I or IV, representing design radii of action of 25 and 150 nautical miles respectively, suffer increasing penalties in effectiveness as the actual operation diverges from the design point. However, systems II and III suffer lower penalties for the same degree of divergence from the design point. If each mission radius of action has equal occurrence probability, the choice of system therefore can be made between II and III; however, since System II displays a lower cost per ton nautical mile at design point, the total penalties for off-design point operation will also be lower, indicating this system to be the optimum choice.

If the various mission radii of action can be assigned varying probabilities of occurrence on the basis of operational experience, the optimum system will be indicated by the minimum total of the product of occurrence probability and mission penalty. By this process, if the probability of encountering 100 and 150 nautical mile mission radii is high, the optimum choice of system would shift to System III or IV depending upon their relative occurrence probabilities.

[redacted]

	I	II	III	IV
I	0	Negl.	5%	15%
II	5%	0	Negl.	10%
III	10%	Negl.	0	5%
IV	15%	5%	Negl.	0

APPROXIMATE PENALTY FOR INCORRECT DECISION

FIGURE THREE

SUMMARY

Military Force Requirements

Military force requirements are shown in Figure Four in terms of the number of helicopters, maintenance personnel, total cost per day and total gallons of fuel required per day. Force requirements are based on a single infantry division, and a scale-up to larger elements should be conducted with caution, unless the same military situation and airlift requirements exist.

Each of the helicopters listed are optimum choices for the given mission radius, and further, the number of aircraft shown are capable, in each situation, of completing the daily airlift in the same total time.

The table also includes the additional helicopters required by operational attrition losses, however these losses are not accounted for in the total costs since the costs of operational attrition are primarily related to the overall support between the Zone of the Interior and the theater of operations.

HOVER CEILING OGE STANDARD DAY (FT)	HELICOPTER TYPE (ALL TURBINE POWERED)	DESIGN RADIUS OF ACTION (N MILES)	OPTIMUM PAYLOAD (TONS)	NUMBER OF SHIPS REQUIRED	NO. OF MAINTENANCE PERSONNEL REQUIRED	TOTAL COST PER DAY ($1000/DAY)	TOTAL FUEL PER DAY (GAL/DAY)	ATTRITION REPLACEMENT SHIPS/MO.
5000	SINGLE ROTOR,	25	3.2	53	558	54.8	27100	1
		50	3.8	64	723	100.5	45000	1
		100	4.2	76	949	225.7	91400	2
		150	3.8	123	1612	440.0	159200	2
	TANDEM ROTOR,	25	4.3	48	533	55.5	24200	1
		50	4.5	59	681	100.5	42600	1
		100	4.5	84	1040	225.7	92000	2
		150	4.3	113	1493	406.5	168600	2
7500	SINGLE ROTOR	25	3.3	52	558	55.5	27900	1
		50	3.6	67	756	100.5	46700	1
		100	3.7	95	1168	231.8	99600	2
		150	3.6	127	1680	443.0	182900	3
	TANDEM ROTOR,	25	4.2	48	542	56.7	26800	1
		50	4.4	60	705	100.5	45800	1
		100	4.4	85	1082	225.7	99000	2
		150	4.0	117	1567	431.0	183700	2
10000	SINGLE ROTOR,	25	3.3	52	573	56.1	30250	1
		50	3.6	67	773	106.6	50000	1
		100	3.7	96	1270	253.8	109150	2
		150	3.6	127	1758	446.0	204200	3
	TANDEM ROTOR,	25	4.0	49	548	58.5	30350	1
		50	4.3	61	720	105.3	52000	1
		100	4.2	88	1120	238.0	113100	2
		150	3.8	124	1680	465.0	216300	2

NOTE: Based on supply requirements of an isolated infantry division in assault of a prepared position. Total tonnage required is 610 tons per day.

FORCE REQUIREMENTS FOR OPTIMUM SYSTEMS

FIGURE FOUR

TABLE OF CONTENTS

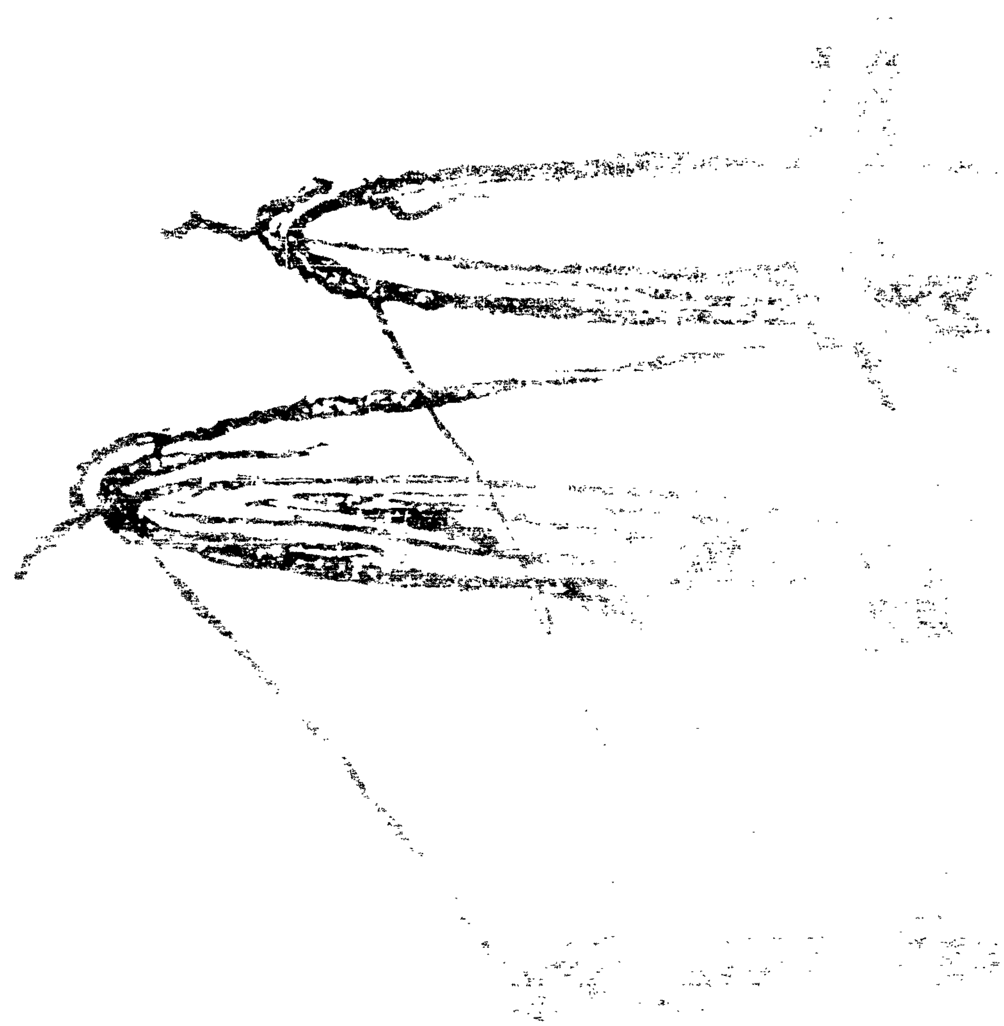
PART ONE APPROACH TO THE PROBLEM

PART TWO COMPONENTS OF THE PROBLEM

PART THREE RESULTS

PART FOUR SUPPORTING DATA

APPROACH TO THE PROBLEM



Chapter I
OUTLINE AND STATEMENT OF THE PROBLEM

A. Statement of the Problem

On 18 February 1954, Contract Nonr 1340 (00) between the office of Naval Research, Air Branch, and Hiller Helicopters was altered in its scope and approach to cover a broad study involving the parametric evaluation of future transport helicopters in Army logistic transport missions.

Specifically, techniques and criteria for evaluating future helicopter types were developed.

No attempt has been made to postulate the pure helicopter as the most efficient short-haul military airlift vehicle nor has any effort been expended to compare the pure helicopter with any other means of transportation.

B. Study Objectives

The primary objectives were to establish quantitative specifications for optimum military helicopter transport systems as a guide to Army procurement planning for the time period 1956 to 1961. In arriving at these objectives, the significant trends of technical, economic and operational parameters on transport effectiveness were developed and evaluated. This evaluation was the result of careful analysis of many interrelated factors which had a direct influence on the system.

A second objective of the study, which by its very nature formed a portion of the primary objective, was to establish a suitable technique for the selection of transport helicopter configurations to meet any given specification. This objective covered the technical aspects of the overall problem and dictated the formulation of generalized design analysis methods which could be used by military procurement personnel in preliminary design performance and weight estimations.[1]

A final objective of the study was to consider pure helicopter types of an advanced design for the procurement time period 1960 to 1970, for the purpose of providing indications of optimum research and development funding during the 1956 to 1961 period.

It is believed that the results of the study have complied completely with the objectives set forth above and that their proper application can provide the military planner with a valuable management tool in establishing Army transport helicopter procurement over the period of the next five years.

C. Basic Assumptions

The detailed analysis and therefore the results of this study are predicated upon certain basic assumptions. These assumptions have all been carefully studied as to their validity and were reviewed in detail during a presentation of the preliminary findings of the research, at a military conference in Washington D. C.[2] The list of these fundamental assumptions is presented below.

1) *Gross Weight Operation* at all times was assumed. The return airlift was considered equal to the outbound Infantry Division supply requirement.

---

[1] HHReport Transport Helicopter Design Analysis Methods 30 November 1955 Submitted under Contract Nonr 1340 (00)

[2] Shredding Session Nonr 1340 (00) June 28 and 29, 1955

2) *Helicopter Fleets Were Assumed Available When and Where Needed* and no consideration was given to the cost of shipment of helicopter fleets from the Zone of the Interior to a theatre of operations.
3) *A Fixed Size of the Army* was assumed, with the result that base establishment and supply costs, and military basic training costs were eliminated from the study.
4) *Helicopter Combat Attrition* was neglected as a component of the problem, however, helicopter operational attrition was taken into consideration.
5) *A Fixed Aircraft Availability* was assumed which gave rise to a variation in flight utilization with payload depending on the time required for loading and unloading.
6) *Generalized Powerplant Performance Characteristics* were assumed for all design configurations in arriving at optimum decisions, and the penalties in effectiveness brought about by the consideration of actual powerplant availability within the study time period were studied and presented.
7) *Detailed Hourly Flight Scheduling* was not considered. Military Combat Element daily support was considered as the basic requirement, and the time of day during which the airlift would be delivered at any specific point was ignored. Loading and unloading times were accounted for, however, in the effectiveness measure.

D. Scope of the Study

1. *Solution Approach*

The initial step in the attack of the problem was to assemble a study group made up of a staff of specialists in the following fields:

a) Helicopter Aerodynamic, Performance and Powerplant analysis
b) Helicopter structural design and weights analysis
c) Aircraft operating cost analysis
d) Aircraft operations analysis
e) Mathematics and computing

The group made a thorough study of all available technical and operational literature pertinent to the problem and then proceeded to outline the necessary areas of investigation. From this effort, there emerged a complete parametric evaluation of the helicopter design, helicopter operational, helicopter cost and Army mission requirement parameters which would have a material effect on the selection of optimum military helicopter transport systems. For this purpose a measure of effectiveness was developed for quantitative comparisons of the various systems.

2. *Areas of Major Concern*

The following areas of investigation were outlined by the group and work on the study has been essentially along the lines of

a) Aerodynamic, Performance and Powerplant Studies
b) Structural and Weight Analysis Studies
c) Direct and Indirect Cost Studies
d) Helicopter Operational Problems
e) Military Logistic Supply Data Analysis

The aerodynamic, powerplant and performance studies have been mainly centered around the development of suitable performance estimating methods on a dimensionless or at least a generalized basis. This has led to the establishment of basic generalized performance equations.

The structural and weight studies have been primarily concerned with the development of a generalized gross weight equation based on suitable statistical weight data permitting the determination of gross weight as a function of all the pertinent design variables.

Such a generalized system of performance and weight analysis techniques was mandatory before any great number of configuration possibilities could be evaluated on a common basis.

The cost studies were undertaken as a requirement of the measure of effectiveness which was selected. Although the costing analysis work has been extremely broad, the major task has been within the area of maintenance cost trends. A comprehensive survey of statistical data was made and a complete maintenance cost analysis technique has been established together with all the other cost items which enter into the problem.

Helicopter operational studies have included the detail problems of cargo loading and unloading, externally carried payload and mission flight plan analysis together with the evaluation of ground support personnel requirements.

The military logistic data involving the operation of the transport helicopter in various situations was also dictated by the selected measure of effectiveness; and consultation with personnel of the U. S. Army in both Procurement and Operational roles, has led to the establishment of certain operational and mission requirements, corresponding to various military support problems.

3. *Procurement Time Schedules*

So many helicopter configurations and helicopter powerplant systems have been proposed over the period of the past two or three years, that in keeping with the objectives as set forth in Section B of this chapter, it became necessary to separate them into two categories. The first category includes those which could be envisioned for production procurement within a five year period commencing in 1956. The second category includes those which would, in all probability, require considerable research and development work before production procurement could be realized. It was felt that this separation would lend more quantitative value to the 1956 to 1961 study and still allow a reasonably accurate qualitative evaluation of optimum helicopter transport systems for the ten year period of 1960 to 1970. In addition, the results of the 1960 to 1970 study give an indication of the areas into which research and development funds might be channeled during the next five years in order to provide optimum helicopter transport systems during the advanced period. The basic study, then, consists of

a) Configurations which could reasonably be expected for production procurement within the next five years augmented by a study of
b) Configurations of an advanced type which might indicate optimum areas for research and development funding in the next five years and point out possible optimum production procurement for the period 1960 to 1970.

4. *Design Possibilities Considered*

Helicopters considered in the basic study were limited to reciprocating and geared gas turbine powerplant types since these represent the only feasible quantity production within the 1956 to 1961 time period. This is the only restriction imposed on the basic study.

Helicopters considered for the 1960 to 1970 time period were assumed to be powered by powerplant types presently considered more radical and less developed; namely, ramjets, pressure jets, and tip mounted turbines.

In Figure I-1 the complete matrix of design possibilities considered is presented. Both single and tandem rotor configurations have been considered through design radii of 25-150 nautical miles with a payload span of from one to five tons for both reciprocating and turbine powerplants. Three degrees of hovering performance have also been included in the basic study matrix. These various possibilities were processed through the measure of effectiveness (see Chapter II) in order to select the optimum systems for the various airlift problems envisioned.

Since various mission radii were also considered, it followed that all helicopter design configurations would be operated at "off-design" conditions as well as at their design point.

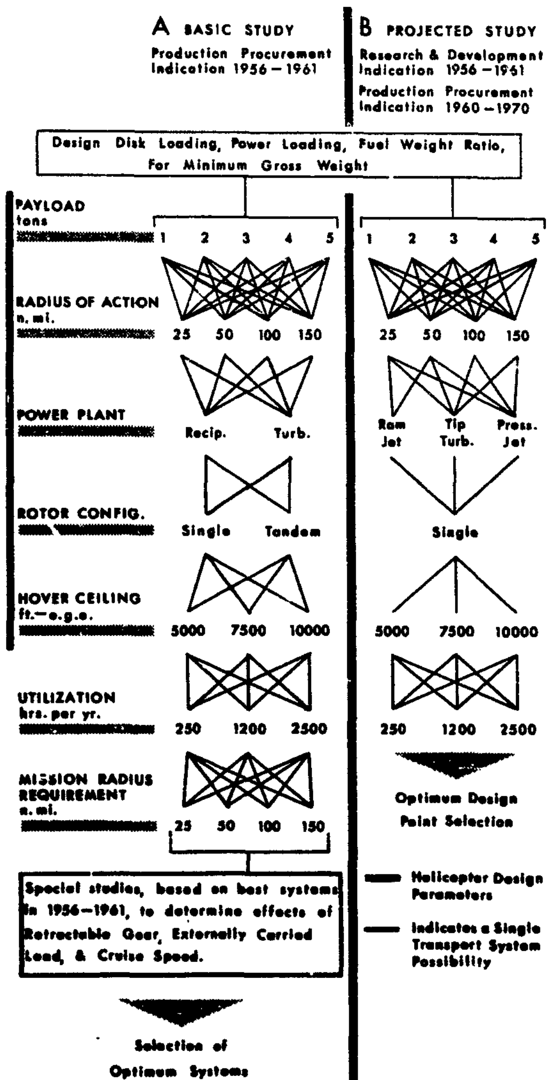
Within the framework of the 1956 to 1961 matrix, additional investigations, augmenting the basic study, were made on the effects of retractable landing gear and externally carried load. These additional investigations were conducted expressly to determine

a) The advisability of retractable landing gear for various payload radius design configurations

and

b) The changes in transport effectiveness for various payload-radius design configurations when the payload was carried externally.

In the projected study, 1960 to 1970, the aspects of rotor tip powerplants were investigated in matrix form to determine optimum design point helicopter configurations for the advanced period. This facilitated a comparison of future possible helicopter types with the optimum design point indications from the 1956 to 1961 matrix.



I-1 SCOPE OF THE STUDY MATRIX

The parameters for all helicopter design configurations, both in the basic and projected study, were selected on the basis of minimum gross weight, and the technique employed for enforcing this criterion is discussed in Chapter IV.

Chapter II
TRANSPORT SYSTEM EFFECTIVENESS

A. Military Logistic Transport

The problem, as set forth in Chapter I, was initially confined to consideration of the supply support problem of Army combat elements. The initial study specifically considered the supply support of an Infantry Division. However, final analysis of the problem has indicated that the resulting optimum decisions are unaffected by the size of the unit requiring support or the support airlift in tons. Military force requirement data resulting from optimum decisions and presented in Chapter IX, represent the support of a single Infantry Division as outlined in Chapter III, but may be linearly scaled for force requirement estimates of larger or smaller combat elements providing the airlift tonnage requirement is linearly scalable.

The problem involved the airlift of supplies, equipment, and personnel from a base Army supply depot to division and regimental supply dump units. No consideration was given, however, to the problem of airlifting supplies, equipment, and troops from a point within an air base complex of the theatre of operations to the Army supply depot or of the initial transportation from the Zone of the Interior to the theatre of operations. Considerations of this type and scope should be made before an efficient and well-organized over-all supply pipe line can be realized.

The problem reduced to a situation which was somewhat analogous to the route structure establishment of a commercial airline, depending on the particular piece of geography involved. The route structure will vary, but remains essentially the same type of operation experienced by the local service airlines. A circuit must be traversed a given number of times per day, depending on the airlift requirements of various branches of this circuit. In a military transport problem, these branches have various cargo requirements which must be fulfilled in accordance with the fleet capability. After initial study, however, it was found that the details of route structure and variations in mission flight plans had negligible effect on the effectiveness measure so that various mission problems could be reduced to the consideration of airlift tons required and the average trip length between the army supply depot and the combat element supply dumps. The major concerns within the problem were, therefore, that of airlift capability or transport work capacity and, a measure of effectiveness which would adequately describe the relationship of all the pertinent parameters.

B. Measure of Effectiveness

For a parametric analysis of this type, it was necessary to devise some criterion which contained the pertinent variables of all components of the problem and which would allow a study of the effects of each of these variables in such a manner as to arrive at optimum combinations of the parameters. This criterion forms the measure of effectiveness, or mathematical model by which the best or optimum system is determined.

The first major problem in establishing this criterion was to determine the relative importance of the various components of the problem in terms of the frame of mind of the military commander or logistics planner. After considerable deliberation, the criterion of transport work capacity per total invested military dollar appeared as the most likely and most pertinent of effectiveness criteria considered. The military commander in a logistic supply situation is most concerned with transport work capacity, defined as ton-miles per hour. However, from the standpoint of the military planner, faced with the economic and budgetary problems of maintaining an adequate defense potential, the consideration of the cost for this work capacity is of equal importance. Furthermore, in any type of transport vehicle, the work capacity increase per unit of cost expended will tend to minimize at some cost value. Tnis merely indicates that, as the work capacity is increased, the expended cost per unit of work capacity increases also, so that at some point consistent with the design "state of the art" of the equipment, an optimum work capacity is obtained. Beyond this optimum point, the work capacity per unit of cost expended will decrease.

After selecting this criterion, it was possible to consider the many components of the problem which required study and analysis. This led to a mathematical expression for the measure of effectiveness in a form which could be most useful in processing the data necessary to evaluate the problem. The mathematical presentation of the measure of effectiveness is presented on the following page, together with an explanation of the general areas of concern.

$$\text{Mission Effectiveness} = \frac{\text{Ton Miles Transported/ Total Hour of Operation}}{\text{Total Military Cost/ Total Hour of Operation}}$$

$$E = \frac{\text{Ton Nautical Miles}}{\text{Total Military Dollar Expended}}$$

The inverted measure takes the form, more commonly accepted, of

$$1/E = \frac{\text{Total Military Cost}}{\text{Ton-Nautical Mile}}$$

For an arbitrary route structure having θ stage lengths, individually designated by j:

$$1/E = \frac{\sum_{j=1}^{\theta} c_j \left(\frac{Tt}{P}\right)_j}{\sum_{j=1}^{\theta} \left(TR\right)_j}$$

Where $C_j =$ Total Military Cost/Hour over the j th stage length

When a single radius or range mission is considered $\theta = 1$

and

$$1/E = \frac{C}{PV_B}$$

To facilitate symbol recognition and understanding of interrelated problem components, this flap should remain unfolded.

EXPANDED EFFECTIVENESS EQUATION

$$\frac{1}{E} = \text{[illegible]}$$

T ---- Airlift requirement ..... tons/day
R ---- Stage distance or radius of action ..... nautical miles
K_P ---- First and spares cost production quantity correction factor
K_{PI} ---- Price index factor
U ---- Aircraft yearly utilization ..... flight hours/year
P_D ---- First and spares cost depreciation period
C_{U_n} ---- Unit first cost of component group n ..... dollars/lb.
R_n ---- Component group weight ratio ..... W_n/W
W ---- Helicopter gross weight ..... lbs.
OHP_n ---- Average overhaul period of component group n
K_{fo} ---- Fuel and oil cost constant ..... dollars/lb.
r_f ---- Fuel weight to gross weight ratio W_f/W over the j th stage length
t_j ---- Flight time over the j th stage length ..... hours
K_{CM} ---- Military maintenance cost correction factor
K_{2M_n} ---- Slope of flight hour cost vs. weight curve for component n

P_j ---- Allowable payload per aircraft per trip ..... tons
K_{IM_n} ---- Intercept of flight hour cost vs. weight curve for component n
K_D ---- Development cost constant
P_W ---- Development cost write-off period ..... years
N_S ---- Number of aircraft procured
ϕ ---- Ratio of empty weight to gross weight ..... W_E/W
R_{EN} ---- Ratio of engine weight to gross weight ..... W_{EN}/W
V_B ---- Block speed in knots
P_s ---- Average service period of flight and maintenance personnel ..... years
U_C ---- Flight crew utilization ..... hours/year
C_{CA} ---- Annual flight crew cost ..... dollars/year
j ---- An index referring to a route structure segment
n ---- An index referring to a given helicopter component group
C_{TM} ---- Maintenance personnel training cost (dollars)
N_M ---- Number of maintenance personnel
C_{TFC} ---- Total flight crew training cost ..... dollars

$\left(\frac{T_t}{P}\right)_j$ Total flight hours/day required to complete the air lift over the j th stage length

Logistic Transport Measure of Effectiveness
(Inverted)

$$\frac{1}{E} = \frac{C}{W_c} = \frac{\text{Total Invested Military Dollar}}{\text{Transport Work Capacity}}$$

$$\frac{1}{E} = \frac{\sum_{j=1}^{e}\left|\left\{\overbrace{\frac{K_P K_{PI}}{UP_D}\sum_{0}^{n}\left[C_{U_n}R_n\left(1+\frac{U}{4\times OHP_n}\right)\right]}^{1}+\overbrace{K_{fo}K_{PI}\left(\frac{r_f}{t_j}\right)}^{2}+\overbrace{K_{C_M}K_{PI}\sum_{0}^{n}\left(K_{2M_n}R_n\right)}^{3}\right\}\overbrace{W\left(\frac{T_t}{P}\right)_j}^{4}+\left\{\overbrace{K_{C_M}K_{PI}\sum_{0}^{n}\left(K_{1M_n}\right)}^{5}+\overbrace{\frac{K_D K_{PI}}{P_W UN_s}\left(\Phi-R_{EN}\right)W^{\epsilon}}^{6}+\overbrace{\frac{K_{PI}}{UP_s}\left(C_{T_{FC}}+K_{C_M}C_{T_M}N_M\right)}^{7}+\overbrace{\frac{K_{PI}}{U_C}C_{C_A}}^{8}\right\}\left(\frac{T_t}{P}\right)_j^{*}\right|}{\sum_{j=1}^{e}\left\{TR\right\}_j}$$

MISSION FUNCTION
ROUTE STRUCTURE SUMMATION
OF DAILY TON-MILE TRANSPORT AIRLIFT REQUIREMENT

1 Summation of Helicopter Component Systems 1st & Spares Cost Depreciation Adjusted to Mid-Point of Study and Corrected for Production Quantity

2 Helicopter Fuel & Oil Cost for a Given Route Structure Stage Length

3 Summation of Helicopter Component Systems Maintenance Costs which are Functions of Gross Weight & corrected for Design Power Loading, Flat Plate Area Ratio, Cruise Speed and Disk Loading. Also Adjusted to Military Cost Level

4 Helicopter Flt. Hrs.* Per Day to Meet the Airlift Requirement over a given Route Structure Stage Length adjusted for "Off Design Point" Payload and Fuel if required

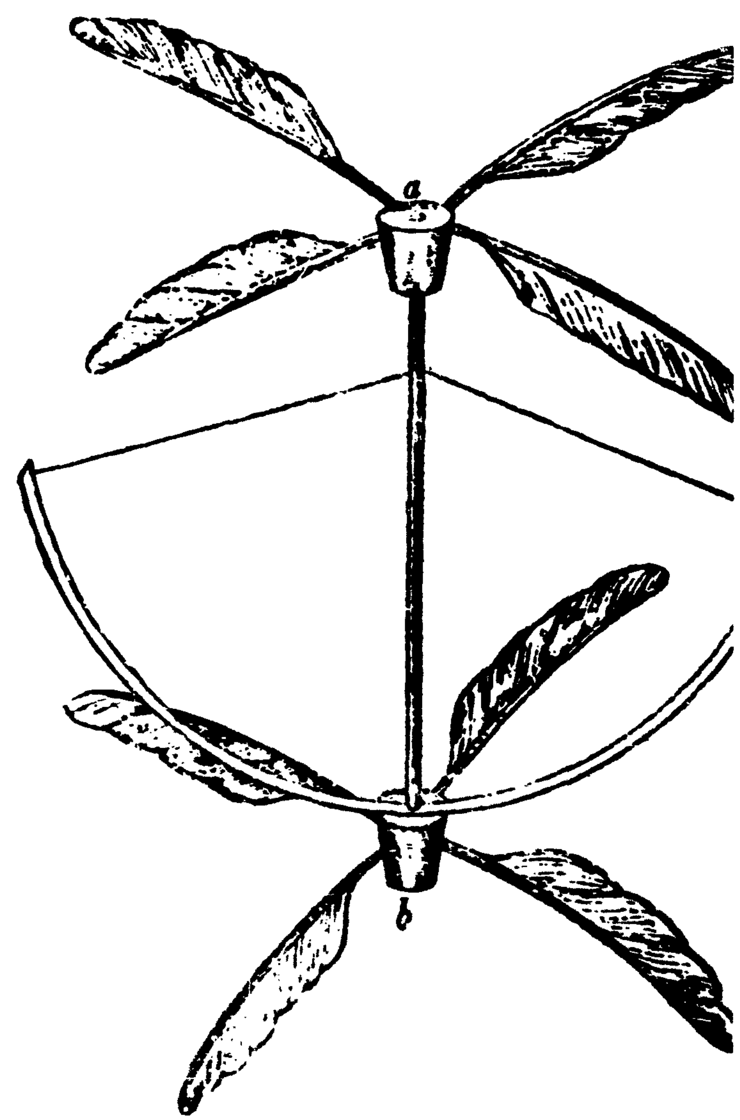
5 Summation of Helicopter Component Systems Maintenance Costs NOT Functions of Gross Weight Adjusted to Military Cost Level

6 Helicopter Development Cost Write-Off

7 Helicopter Flight Crew & Maintenance Ground Crew Training Cost Write-Off

8 Helicopter Flight Crew Operations Cost

COMPONENTS OF THE PROBLEM



Chapter III
COMBAT ELEMENT SUPPORT REQUIREMENTS

A. Typical Support Problem

A survey of Army organization and supply data was undertaken to allow the establishment of a typical military supply problem for an infantry division. Based on this hypothetical problem, the measure of effectiveness was to be employed to evaluate the various design configurations in their performance of the stipulated mission, that of complete logistic support of the division. The objectives of the survey therefore, were to compile the information necessary to define a hypothetical tactical situation with a logical deployment of units and having this, to determine the supply requirements of each unit and the transportation network required to fully maintain it. No attempt was made to arrive at the optimum transportation network since the primary objective in establishing the supply problem was to provide only reasonable estimates of anticipated supply requirements and route structure.

Consideration in the survey was therefore given to the following topics which are further discussed in the following paragraphs:

1) Geography
2) Infantry division organization, equipment and tactics
3) Logistics

The terrain over which the division supply network was to operate was laid out without reference to actual geography. This allowed the supposition of obstacles in trip length and flight altitudes which would strain the supply lines and eliminated a time consuming search for actual maps that would indicate the detailed terrain features.

The tactical situation was laid out on the hypothetical terrain, as shown in Figure III-1, and the supply network was superimposed, showing the supply routes from an Army supply depot through division to regimental supply dumps. Regimental supply dumps were assumed to be the forward limits of helicopter operation since in the opinion of U. S. Army personnel, fleets of cargo helicopters would not be employed continuously in forward areas under enemy observation.

The tactical situation was assumed to be representative of World War II action, however, the divisional front was expanded to a distance of 30 nautical miles and the combat zone depth to a distance of approximately 75 nautical miles in a further effort to burden the supply system. The classic order of battle was assumed in which, for a triangular organization, one unit attacks, one unit holds, and one unit is in reserve. Referring to Figure III-1 it may be seen that the attacking regiment reenforced by the Reconnaissance Company, two battalions of Artillery and an Engineer Battalion, draw supplies from point E. The holding regiment, reenforced by Division Artillery draw supplies from Point D. A portion of the reserve forces, comprised of two infantry battalions, regimental headquarters units and units of division headquarters draw supplies from point C. The remaining division elements, including Ordnance, Engineer, Medical, Quartermaster, Tank battalion, Anti-aircraft artillery, and one infantry battalion, draw supplies from point B.

The following additional assumptions were made prior to estimating the support requirements of the division:

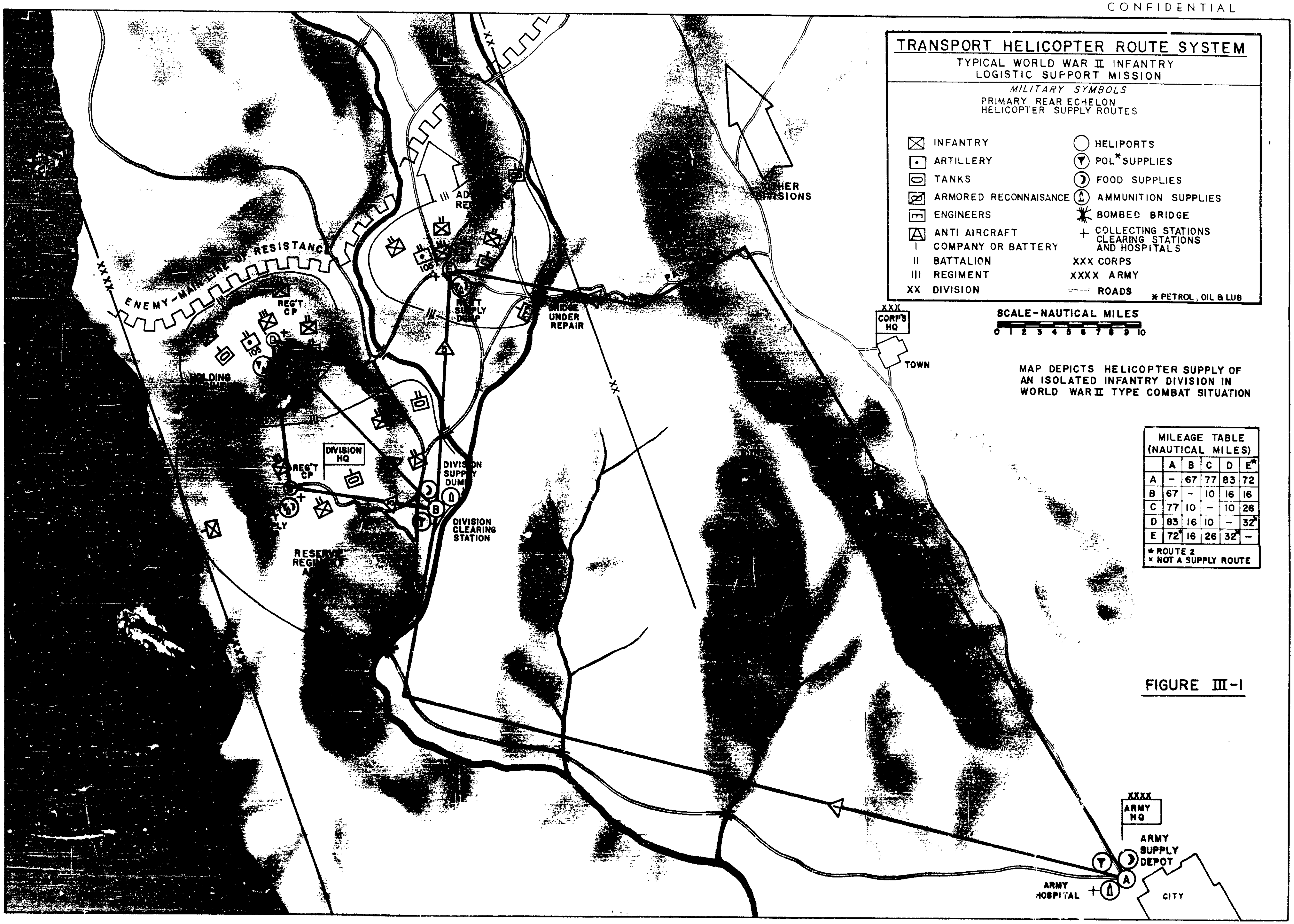
1) Personnel:
   The unit to be supplied was an infantry division with personnel and equipment per FM 101-10[1], to be maintained at T/O strength of 18180 men.
2) Operation:
   a. The division was attacking a prepared position as a part of an advancing Army front.
   b. All supplies including mail were to be transported by the helicopter supply fleet from the Army Supply Depot, (A), to the combat elements. Water was to be obtained at point B.
   c. Evacuation of casualties and transportation of replacement troops were assumed to be accomplished by the helicopter fleet.
3) Supplies:
   a. It was assumed that initial supply was in the field.
   b. Resupply was estimated from the daily requirements of an infantry division in the assault of a prepared position.

From the preceding data and assumptions, the tonnage requirement for the hypothetical situation was analyzed and is tabulated in Figure III-2. For each class of supply the figure shows the point of origin, destination, and tons required per day. In the lower table, the units supplied and the total number of personnel within these units are shown.

[1] FM 101-10 *Staff Officers Manual*, Organizational, Technical and Logistical Data, July 1953.

Definition	Type of Supply	Point of Origin	Supply Required at Point Tons/Day B	C	D	E
Food	Class I	A	11.34	9.32	13.79	14.40
Equipment	Class II & IV	A	14.71	9.63	7.24	13.62
Petrol, Oil Lubrication	Class III	A	37.15	13.25	10.37	11.23
Ammunition	Class V	A	13.0	30.0	173.0	187.0
Mail		A	.59	.49	.72	.75
Water		B		6.92	10.26	10.70
Personnel		A	1.64	2.10	3.47	3.70
		Totals	78.43	71.72	218.85	241.40
		NUMBER OF TROOPS	4224	3468	5133	5355
		UNITS SUPPLIED	Inf. Bn (1) Q.M. Co. Repl. Co. Ord. Bn Tank Bn Med. Bn Engr. Bn (Partial) Div. Arty. (Partial) Band	Div. H.Q. H.Q. Co. Med. Det. Signal Co. MP Co. Inf. Regt. (Less 1 Bn)	Inf. Regt. Div. Arty. (Partial)	Inf. Regt. Engr. Bn (Partial) Recon Co. Div. Arty. (Partial)

III-2 TABLE OF AIRLIFT REQUIREMENTS



MILEAGE TABLE (NAUTICAL MILES)	A	B	C	D	E*
A	–	67	77	83	72
B	67	–	10	16	16
C	77	10	–	10	26
D	83	16	10	–	32×
E	72*	16	26	32×	–

* ROUTE 2
× NOT A SUPPLY ROUTE

FIGURE III-1

CHAPTER III — COMBAT ELEMENT SUPPORT REQUIREMENTS

Data from Figure III-2 combined with the mileage data from Figure III-1, give the daily ton mile transport requirement for a given route segment. This quantity, the product of the tonnage required at a given point and the mileage flown on the transport mission, is expressed symbolically in the measure of effectiveness as:

$$T_j R_j$$

in which j refers to a given route segment.

B. Additional Mission Considerations

Helicopter operational factors and techniques are discussed in Chapter V. Consistent with the assumptions made therein, the measure of effectiveness was calculated for the operation of fleets of the various design configurations in performance of the division supply problem.

The values of cost per ton mile obtained by the detailed consideration of routing, various cruise altitudes and various route segment distances were found to be negligibly different than the cost per ton mile calculated for an average radius mission or average ton-mile requirement at a single altitude. Consistent with this, a series of standard missions were established within the operational matrix to show a range of ton mile mission requirements.

Chapter IV
HELICOPTER DESIGN SELECTIONS

A. Scope of the Design Selection

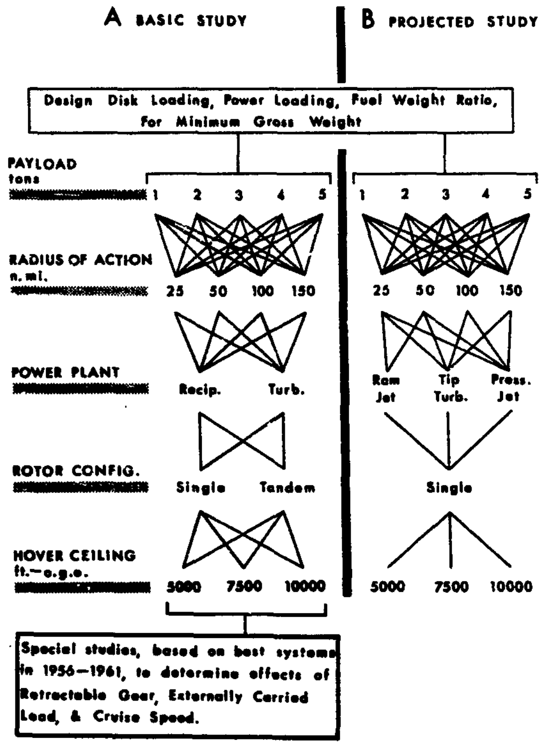
The performance and design configuration matrix shown in Figure IV-1 illustrates the various combinations for which minimum gross weight helicopter design parameters were analytically determined.

The parametric analysis technique which was developed for use in establishing the minimum gross weight designs for each mission-configuration combination in the matrix is described briefly in Section D of this chapter, and in greater detail in the Design Analysis Methods report.[1]

It should be noted that this basic matrix considered only helicopters with *fixed landing gear,* and with *payload carried internally.* Furthermore, the general analysis was based on established "state of the art" cruise speed limited by the onset of rotor tip compressibility drag rise. Special analyses of the effects of retractable landing gear, externally carried payload, and higher cruise speeds (based on the pure assumption that compressibility drag divergence may be tolerated, provided enough power is available, and is not a rotor roughness limit) were superimposed upon certain representative designs which evolved from the basic 1956 to 1961 study. These special studies are discussed in greater detail in sections which follow.

B. Transport Helicopter Layout and Design Studies

To insure compatibility between cargo, fuselage, and rotor dimensions, a series of preliminary design layouts were made for each payload, powerplant type, and rotor configuration considered in the 1956 to 1961 study. Fuselage dimensions and arrangements were based on either troop seat requirements or the dimensions of certain military items in the February 1953 AFF Equipment Data book.



IV-1 DESIGN PARAMETER MATRIX

[1] *Transport Helicopter Design Analysis Methods.* H. H. Report No. 473.6; 30 November 1955.

CHAPTER IV — HELICOPTER DESIGN SELECTIONS

In all cases, compatibility and reasonable appearance were achieved without deviation from established design practice upon which the aerodynamic and weight statistics are implicitly based in this study. The various typical cargo types and dimensions for payload categories of one, three, and five tons are summarized in Figure IV-2.

From the layout design studies, it was concluded that rear ramp loading is most feasible for payload categories of one and two tons, whereas for three to five ton payload categories either rear or front ramp loading could be achieved in a reasonable design.

C. Aerodynamics, Powerplant, and Weight Critique

In a study of this nature, the number of variables must be kept to a reasonable minimum, and time-consuming attention to minute detail must be avoided wherever possible. While the "state of the art" of helicopter design has not crystallized to the extent that it has in fixed-wing aircraft, there are nevertheless, certain practical design limits imposed by aerodynamic, structural and manufacturing technology. Other limits, some manifest and some subtle, are imposed by the general purpose, or mission, for which the machine is to be designed.

Payload Category (Tons)	Typical Military Cargo	Cargo Weight (lbs.)	Width x Height x Length (Ft.)
1	10 Troops (5 seats each side, facing inward)	2000	6 x 5 x 8
	T-66B2 Rocket Launcher	1270	5.7 x 3.9 x 10.9
	M1A1 75mm Pack Howitzer	1480	4.3 x 3.1 x 10.6
	¼ Ton Trailer	1050	4.7 x 3.4 x 9.1
3	30 Troops (15 seats each side, facing inward)	6000	6 x 5 x 16
	M29 Cargo Carrier	5277	6 x 6 x 10.5
	M2A1 40mm Gun	5850	6 x 6.6 x 22.0
	M2A2 105mm Howitzer	5130	5.1 x 5.2 x 17.6
	Two M1A1 75mm Pack Howitzers Ammo.	2960	4.3 x 3.1 x 21.2
	Generator, Portable	5280	3.0 x 5.4 x 8.1
	M-38 ¼ Ton Truck (Jeep)	3425	4.8 x 4.6 x 11.1
5	50 Troops (13 double-seat rows, each side)	10000	9 x 6 x 40
	T-78 76mm Gun	8000	5.9 x 5 x 17.8
	T-8 90mm Gun	7570	8.3 x 5.5 x 23.1
	M-10 Ammo. Trailer	6250	7.2 x 4.9 x 12.5
	M-37 ¾ Ton Truck	7800	6.2 x 5.3 x 15.4
	Four 75mm Pack Howitzers Ammo. (2 ea. side)	5920	8.6 x 3.1 x 21.2
	Two M-38 ¼ Ton Trucks	6850	4.8 x 4.6 x 22.2

IV-2 CARGO DIMENSIONAL DATA

Therefore, in the initial phase of this study, a concerted effort was made to assign *fixed values*, or *standard variations* to as many design parameters, performance specifications, powerplant characteristics, and general dimensional relationships, as could be justified either by statistical trends extrapolated into the near future, or by preliminary aerodynamic, structural, and weight analyses. It is to be expected that the ever-advancing "state of the art" may prove some of the assumptions in this study to be conservative. However, it is firmly believed that these expected improvements would in no case invalidate the *comparative* results, nor cause an erroneous indication of the type and size of helicopter which should be procured for a given transport mission, *within the time period covered.*

Some discussion of the aerodynamic, powerplant, and structural weight assumptions was given in the Interim Report[1] of this contract. These considerations are treated in greater detail in the following paragraphs, and in Appendices B, C, and D.

Rotor tip speed for all helicopters in this study was fixed at 700 ft/sec. This afforded the best compromise between high rotor and transmission weights at lower tip speeds, and high installed power requirements at higher tip speeds, the latter being magnified by the onset of rotor tip compressibility drag rise at high forward speeds. A detailed discussion of rotor tip compressibility phenomenon is given in Appendix B.

Rotor blade loading was, with one exception, set at *87.3* lb/ft². This value in combination with the selected tip speed gives a reasonably high rotor lift-drag ratio without danger of retreating blade tip stall at forward speeds up to 120 knots at 5000 ft. altitude. The one exception mentioned involved a special study to investigate the effects of higher cruise speeds, up to 130 knots, as discussed in the following paragraph.

Cruise speeds for all design configurations in the initial study were selected as the speed for maximum miles per pound of fuel, up to an assumed tip compressibility drag rise limit of *120 knots at sea level,* decreasing to a limit of *111 knots at 5000 feet.* These limits correspond to an assumed tip drag divergence Mach number of .81, for the fixed tip speed of 700 ft/sec. Recent indications[2] that tip compressibility is not a rotor roughness limit and that tip Mach numbers approaching .9 or 1.0 can be tolerated provided sufficient power is available, led to the inclusion in this study of a special analysis of higher cruise speeds up to 130 knots at 5000 feet. This analysis was made for single rotor, geared gas turbine helicopters only, as a representative example. As shown in Appendix B, Figure B-2, the retreating tip stall limit at 5000 ft. and 130 knots forced a reduction in blade loading to 73.7 lbs/ft² for this special case.

Rotor blade airfoil section profile drag coefficient was assigned a standard variation, $c_d = .009 + .3\alpha_r^2$ where α_r is the mean blade angle of attack. This drag polar is representative of symmetrical airfoil sections with thickness ratios of 12 to 18%. The use of other airfoils, such as the "64" series, would have negligible overall effect on the power and weight, and hence on the comparative results.

Anti-Torque tail rotor dimensions for single rotor geared-drive helicopters were set by the assumption of a tail rotor to main rotor radius

[1] *Military Transport Helicopter Systems, Interim Report, Aims and Scope.* H. H. Report No. 350.0; 1 February 1955.

[2] Unofficial flight test reports of Bell Aircraft Co. and Sikorsky Division of United Aircraft Corp.

ratio of *.18*. The distance between main and tail rotor hubs was set at *1.23* times the main rotor radius, giving a constant clearance ratio between main and tail rotor blade tips of *.05* times the main rotor radius. Tail rotor tip speed and hovering blade loading were assumed the same as for the main rotor. From these assumptions emerged a standard variation of tail rotor power in percent of total power required, versus forward speed. This variation, showing a tail rotor power of 7.7% in hovering and 5% at 120 knots airspeed, is shown in Appendix B, Figure B-3.

Tail rotor dimensions for single rotor tip-drive helicopters were set by the assumption of tail rotor to main rotor radius ratio of *.12*. The distance between hubs was set at *.75* times the main rotor radius. These assumptions provided, in all cases, adequate yaw control, in accordance with the requirements given in Military Specification MIL-H-8501. A 3% tail rotor drive system power loss was assumed. From these assumptions emerged a standard tail rotor power variation with forward speed, included in Appendix B, Figure B-3, showing a total tail rotor power loss of 2.3% of total power in hover, 4.5% at 60 knots, and 3.3% at 120 knots.

Tandem rotor dimensions, tip speed, and blade loading were chosen identical with single rotor configurations, with the additional assumption that the intermeshing rear rotor overlaps the front rotor by .6 times the rotor radius, giving a total distance between hubs of 1.4 times the rotor radius. Also, the vertical displacement of the rear rotor above the front rotor was fixed at .1 times the rotor radius.

These relationships, together with the assumption that the working disk area of the overlapped rotors is the projected area, combined to give an induced power interference correction factor for tandem rotors in the order of 1.65 times the isolated single rotor induced power, at all speeds above best climb speed. More detailed discussions of the considerations involved in the tandem rotor analysis are given in Appendices B and D.

Transmission and drive system power loss was assumed to be *3%* of total transmitted power, for single rotor shaft-drive helicopters, and *4%* of total transmitted power for tandem rotor shaft-drive helicopters. The higher gear loss for the tandems was assumed to account for the additional main transmission and intermediate or right angle gear boxes required.

Twin engine installations were assumed for all shaft-powered helicopters in this study.

Reciprocating engine powerplants were assumed to be radial, aircooled types supercharged to 5000 ft. by single speed gear-driven superchargers and were assigned standard generalized characteristics based on statistics for several operational types[1]. Cooling power was assumed to be 5% of normal rated horsepower at sea level, decreasing with altitude directly as the density ratio.

Geared gas turbine engines were assigned standard characteristics based on a mean fairing of statistical data for engines ranging in size from the Continental Artouste to the Allison T-56.

Tip turbojet engines were assigned standard characteristics based on Packard Motor Car Company estimates[2].

Tip ramjet engines were assigned standard

[1] *Generalized Powerplant Characteristics for Reciprocating Engines*. Douglas Aircraft Co., 14 October 1947.

[2] *Helicopter Tip Turbojet Brochure*, Packard Motor Co., Aircraft Engr. Div. Report No. 7JE-103, 27 September 1954.

characteristics based on Hiller[1] and Marquardt unpublished data.

Pressure jet engine characteristics[2] were based on a pressure ratio of 2.5, the air supply provided by a special centrifugal compressor, driven by geared gas turbines with characteristic identical to those used for direct shaft power. Pressure jet power was set to permit cruising without tip burning, and full power with tip burning up to 3000° F tip burner temperature.

Powerplant specific weights for all types were based on the Thermal Research study[3] modified and augmented where necessary by Hiller and other source data on ramjets, tip turbojets, and pressure jets. Charts showing power and SFC variation for these various types of powerplant are presented in Appendix C, Figures C-1 and C-2.

***Equivalent parasite drag flat plate area* (A_π) for the helicopters in the general study, having fixed landing gear and payload carried internally, was based on the statistical variation**

$$A_\pi = .33W^{.47}$$

established by the Thermal Research study[4]. This variation agrees remarkably well, for gross weights above 5000 lbs., with drag estimates made by the study group for a wide variety of design configurations. For the special studies of the effects of retractable landing gear and external load, this basic expression for A_π was altered, through a general analysis of landing gear drag to be subtracted, and external load drag to be added, giving the following modified expressions:

$$A_\pi \text{ (retractable gear)} = .391\, W^{.42}$$

$$A_\pi \text{ (external load, fixed gear)} = .495\, W^{.47}$$

These relationships are plotted in Appendix B, Figure B-5.

Disk loading and power loading:

These two interrelated design parameters remained in the analysis as primary variables which could not be arbitrarily assigned fixed values, and which would have major influence on the empty weight, gross weight, operating cost, and final measure of effectiveness. Preliminary analysis indicated that disk loadings for minimum gross weight might vary anywhere from 2 lbs/ft^2 to 10 lb/ft^2, dependent upon the design radius of action, payload, and power loading. This range of disk loadings was therefore covered in the parametric analysis. Although disk loadings greater than 8 or 10 would in all probability result in power-off autorotational descent rates which are intolerably high, this limitation was not recognized since all helicopters in this study are assumed to have multi-engine reliability.

Of the various performance criteria which could have been used to establish the design power loading (for example: vertical or maximum rate of climb; top speed; ability to maintain some minimum performance with one engine out; service ceiling; or hover ceiling) hover ceiling out of ground effect, on a standard day, using normal rated power, proved to be the least unwieldy, and was therefore chosen as the best common performance basis for all design configurations. Given a fixed tip speed, blade loading, and blade section drag, the hover power required de-

[1] *Proposal For the Improvement of the Ramjet Engine for Helicopter Propulsion;* Hiller Report No. 545.3; 30 November 1954.

[2] *Pressure Jet Powerplant Considerations;* Appended to H. H. Report No. 473.6, Transport Helicopter Design Analysis Methods, 30 November 1955.

[3] *Helicopter Propulsion System Study*, Thermal Research and Engineering Corporation, Conshohocken, Pa.; September 1952.

[4] Ibid.

pends only on disk loading and air density, thus the required installed power is directly related to, and determined by, the disk loading and hover ceiling.

Helicopter empty weight analyses were, of necessity, statistical in nature, based on data for many actual helicopters. The data was obtained from the previously referenced *Helicopter Propulsion System Study* with some modifications made necessary by the extrapolation required by the scope of gross weights encompassed in the present study. In addition, some of the basic data used by the Thermal Research and Engineering Corp. in preparing the report was obtained and reprocessed by the study group in order to obtain a greater degree of correlation with latest trends in helicopter design "state of the art". Other weight data, particularly that for tip power plants, were obtained from other sources and analysis as mentioned previously.

Since the statistical weight data were derived from helicopters built prior to 1952 there is some question as to the validity of extrapolation. The largest helicopter represented by the statistical weight data was a 35000 lbs. gross weight machine, however, gross weights to 100,000 lbs. were encompassed by the present study. The larger gross weight design configurations were penalized to a slight extent by the assumption of a fixed percentage of gross weight for certain items of empty weight. Whereas the trend of the "item" weight vs. gross weight may be typical for lower gross weight machines, in some cases the "item" is essentially a fixed weight for a given mission type regardless of gross weight. For example, communications equipment was assumed to have a fixed weight for all configurations. The error incurred in empty weight by using the statistical trends for the weight of other items of the same nature would be negligible.

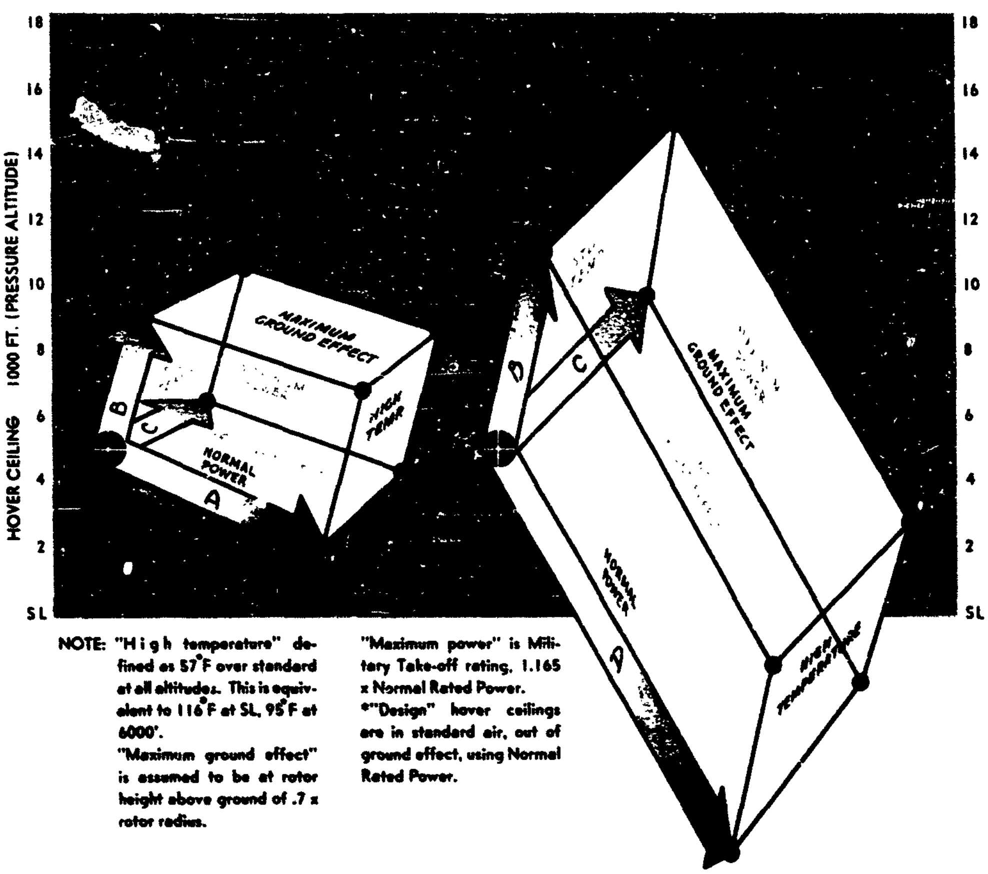
A second possible source of error inherent in the statistical weight data was due to the criterion of hover ceiling. As mentioned previously the design power loadings were determined by the power required to hover at various altitudes. In no case however, was the design hover ceiling for a configuration less than 5000 ft. Consequently, power plant and drives weight for a given gross weight was considerably higher for design configurations included in the study than for machines represented by the statistical data, most of which have marginal hovering performance. By increasing the power plant and drives weight in this manner, the probability was ignored that transmissions and drives specific weights might decrease simply due to design advantages of larger size, or to improved materials and manufacturing techniques.

Considering the sources of possible error, the only available means of estimating the accuracy of the weight analysis was that of comparison with recent designs. However, obtaining pertinent weight and performance data for recent transport designs is complicated first, by proprietary restrictions imposed by the manufacturers, and secondly, if any data is obtained, by ignorance of the stage of development which it represents. Since performance and weight data is known to vary considerably in the progress from preliminary design to operational status, the comparisons were made with reservations. H-37 weight data made available to the contractor plus unofficial but reliable estimates of the operational characteristics, were used to estimate the design gross weight by methods of this study. For this single rotor helicopter, it was found that the predicted gross weight was within 4% of the published design weight. A similar comparison was made for tandem rotor helicopters by comparing the YH-16A gross weight with that predicted by the study. Although data on the YH-16A did not include hover ceiling, this was estimated from the published power loading, and it was determined that the published gross weight and the predicted value were in close agreement.

Further detail on the methods of weight analysis and the assumptions made are presented in Appendix D.

Hover ceiling:

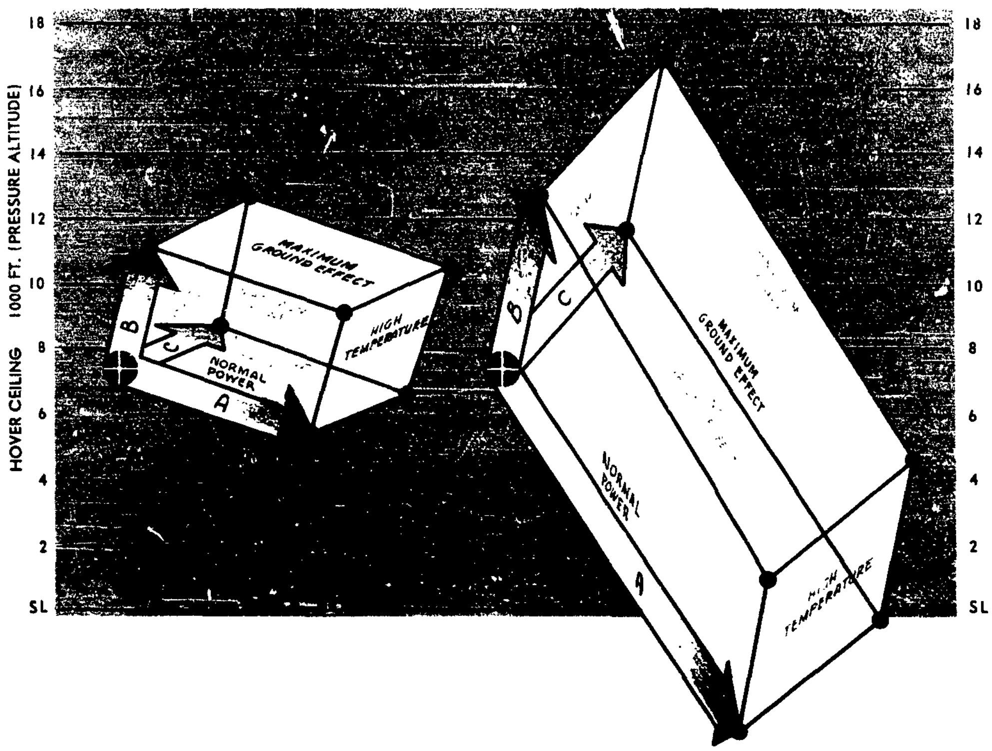
The selection of hover ceiling for a military transport helicopter presents a dilemma for which there is no simple answer. Recent military helicopter specifications have called for a hover ceiling out of ground effect of 6000 feet at 95°F air temperature (57°F over standard temperature at that altitude) using normal rated power. This requirement imposes

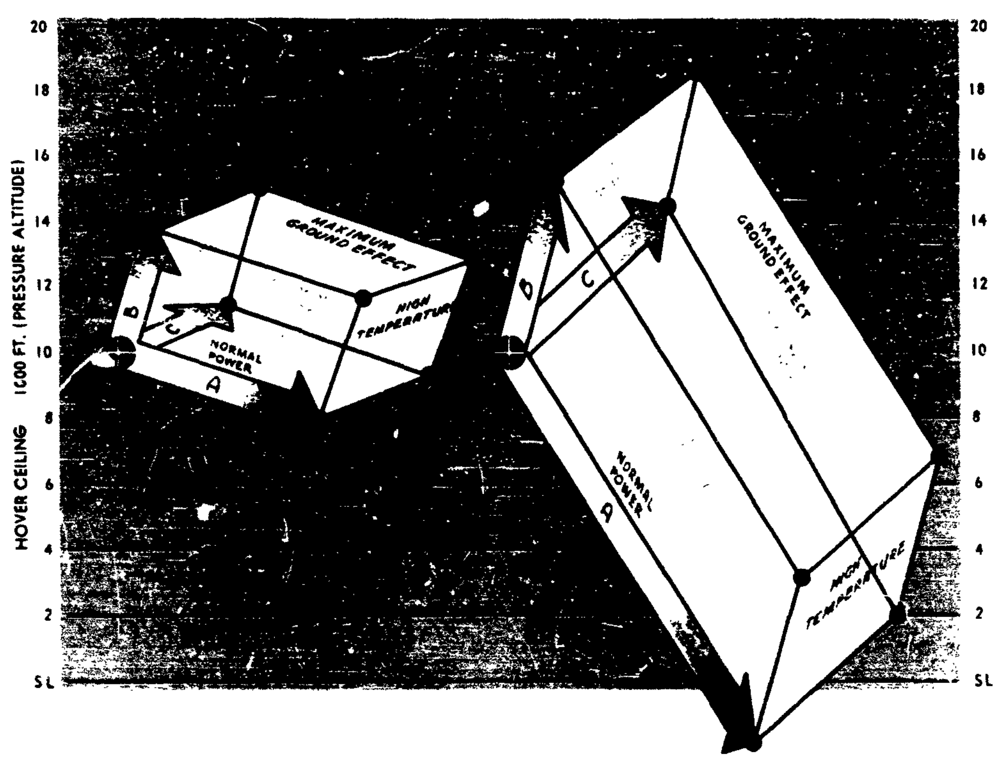


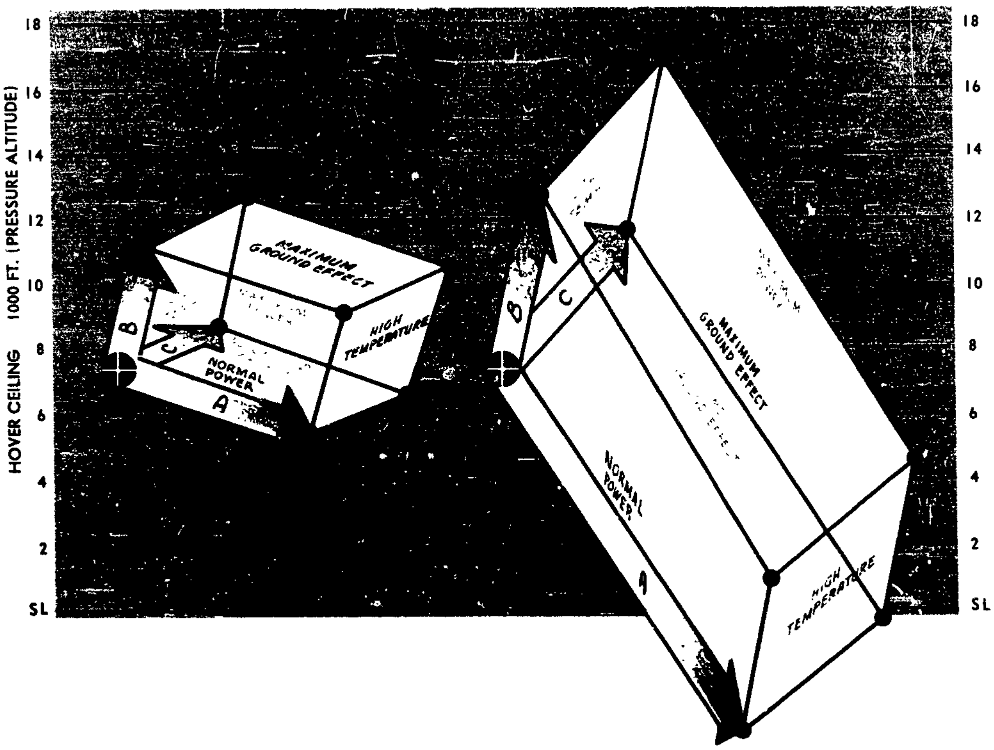
NOTE: "High temperature" defined as 57°F over standard at all altitudes. This is equivalent to 116°F at SL, 95°F at 6000'.
"Maximum ground effect" is assumed to be at rotor height above ground of .7 x rotor radius.
"Maximum power" is Military Take-off rating, 1.165 x Normal Rated Power.
*"Design" hover ceilings are in standard air, out of ground effect, using Normal Rated Power.

IV-3 EFFECT OF AIR TEMPERATURE, GROUND EFFECT AND POWER ON HOVER CEILING WITH RECIPROCATING AND GEARED GAS TURBINE ENGINES

CHAPTER IV — HELICOPTER DESIGN SELECTIONS






HOVER CEILING 1000 FT. (PRESSURE ALTITUDE)
18
16
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12
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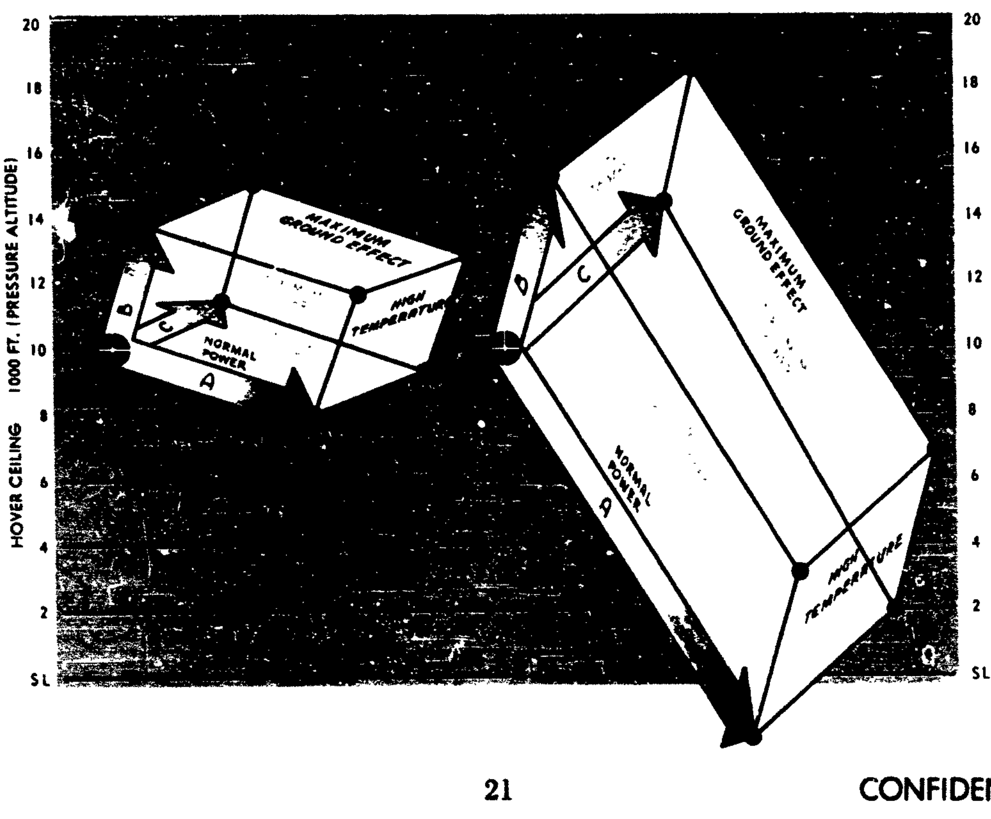
SL
MAXIMUM GROUND EFFECT
HIGH TEMPERATURE
NORMAL POWER
A
B
C


HOVER CEILING 1000 FT. (PRESSURE ALTITUDE)
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2
SL
MAXIMUM GROUND EFFECT
HIGH TEMPERATURE
NORMAL POWER
A
B
C

a very severe power loss on all jet-type engines, as compared with reciprocating engines. Specifically, geared gas turbines, tip turbojets, ramjets, and pressure jets suffer losses *up to 30%* of normal rated power on such a "hot day", as compared to an approximate *5% loss* for reciprocating engines.

A logical approach to the selection of a military hover ceiling requirement would be to perform a thorough climatic and geographic analysis on a global basis, to develop the probable percent occurrence of extreme temperatures at high altitudes. From such an analysis, a standard requirement could be determined which would provide satisfactory operation under the greater percentage of probable conditions. Such a study, however, was outside of the scope of this investigation.

In the considered opinion of the study group the use of *maximum military take-off power* (1.165 times the normal rated power, for all powerplants in this study) should be permitted for hover performance on a "hot day", just as maximum power is used during overloaded take-off and emergency conditions in fixed-wing aircraft operation. It also seems reasonable to assume that most, if not all of the hovering time for these large transport helicopter types would occur within the "ground effect", during take-off and landing. It therefore seems unnecessary to design for extreme conditions, the occurrence of which would be the exception rather than the rule, in the transport operation in which hovering time must be kept to a minimum for maximum effectiveness. Because there is no simple solution to this problem, however, the scope of the design selection matrix (see Figure IV-1, Section A of this chapter) included standard day design hover ceilings out of ground effect, with normal rated power, of *5000* feet, *7500* feet, and *10000* feet. The deviations from these design ceilings when: a) *temperature increases;* b) *ground effect is used,* and c) *maximum power is used,* are illustrated graphically in Figure IV-3. As noted previously, measures of effectiveness were calculated for each of these design conditions for all configurations. The comparative penalty in measure of effectiveness to be paid in return for higher hover ceiling is shown in Chapter VII.

D. Selection of Design Parameters

The development of a suitable technique for the rapid and reasonably accurate estimation of minimum gross weight, and design parameters corresponding thereto, was a necessary work item, and a major task within the study. The method which was developed has become known at Hiller as the *R_F Graphical Method of Parametric Analysis.*[1] The ratio R_F, defined as the ratio of fuel weight to gross weight, provides the basic link between the aerodynamic and weight requirements. Given a specified *payload, radius-of-action,* and *hover ceiling,* the aerodynamic characteristics determine the variation in *required* R_F with gross weight and disk loading, while the empty weight breakdown and payload determine the variation in *available* R_F with gross weight and disk loading.

The method of analysis and graphical solution is discussed in detail in Appendix E and illustrated in Figure IV-4, from which it can be seen that *singular solutions* for gross weight, R_F, and disk loading are obtained from the intersections of the aerodynamic *required* R_F curves with the weight *available* R_F curves, plotted versus gross weight. Each

[1] *The "R_F" Graphical Method of Parametric Analysis for the Development of Optimum Preliminary Design Aircraft.* H.H. Report No. 473.8; 21 October, 1955.

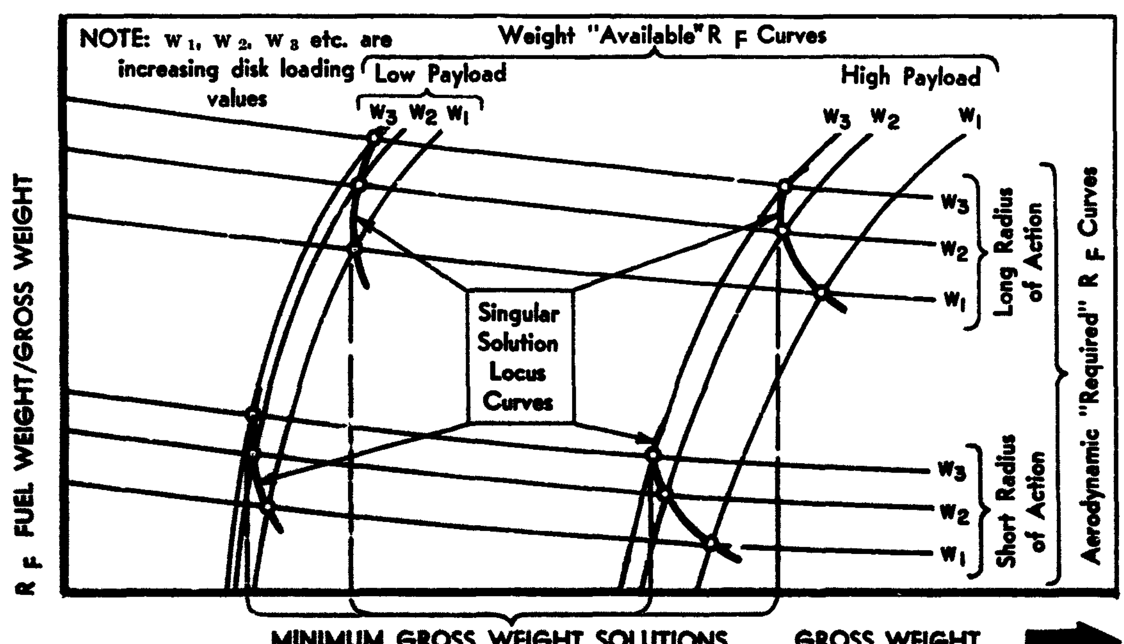
CHAPTER IV — HELICOPTER DESIGN SELECTIONS

intersection represents a specific helicopter design, having a given design payload, design radius of action, design hover ceiling, and design disk loading. By repeating this graphical solution procedure for several disk loadings (w) a locus curve of singular solutions is formed, as illustrated. Each locus curve so obtained indicates a *minimum gross weight* at a particular disk loading and R $_F$, and it is these minimum points which establish the parameters for each design type, payload, radius of action, and hover ceiling.

E. Example Design Characteristics Chart

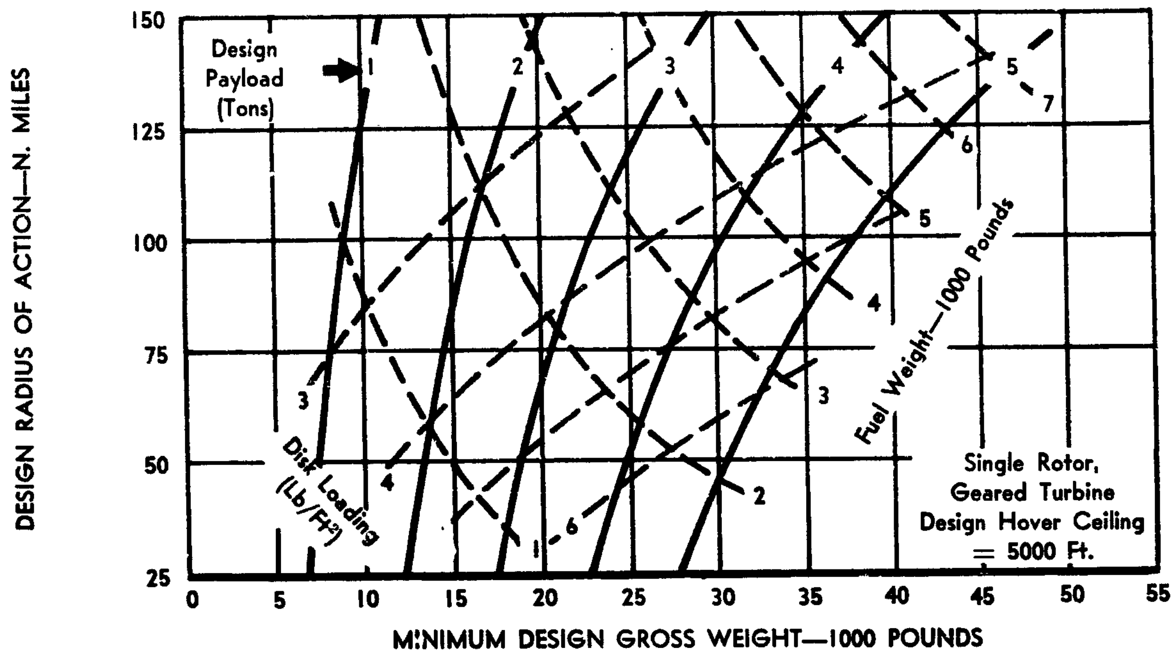
For each helicopter design type within the study matrix (Figure IV-1), characteristics charts were constructed from the singular solutions which were obtained by the method outlined in the preceding paragraphs. These charts show the variation of minimum gross weight with radius of action for payloads of 1 to 5 tons, and include curves for the corresponding disk loadings and fuel weights, as illustrated by the examples in Figure IV-5. These two examples were chosen to compare single rotor and tandem rotor helicopters, each powered by geared gas turbine engines, and each having a design hover ceiling of 5000 feet on a standard day, out of ground effect, using normal rated power. The design characteristics charts for the remaining rotor configurations, powerplant types and hover ceilings considered in this study may be found in Appendix F.

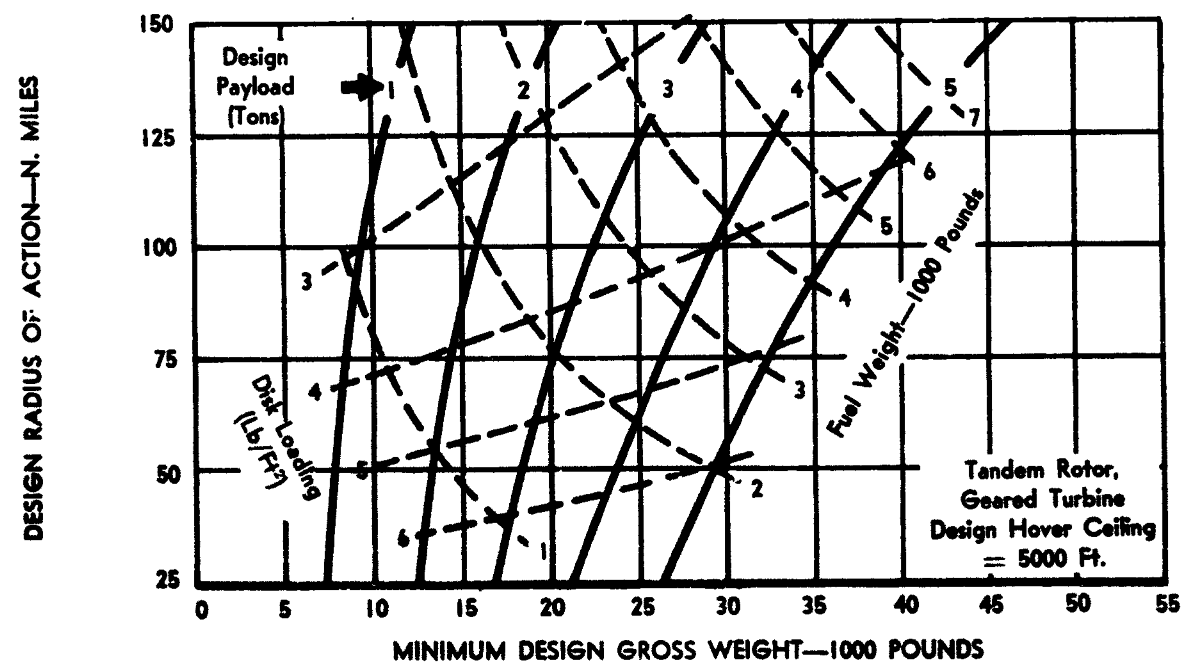
It may be seen from Figure IV-5 that for design payloads of one ton, the gross weights for single rotor helicopters are less than those for tandem rotor helicopters, whereas for pay-



IV-4 GRAPHICAL SOLUTION FOR MINIMUM GROSS WEIGHT AND DESIGN PARAMETERS loads of two and three tons the gross weights are essentially the same for both types, and for payloads of four and five tons, the tandem rotor helicopters become increasingly lighter than single rotor helicopters. The gross weight at which the cross-over occurs, beyond which tandem machines are lighter, appears to be between 15000 and 20000 pounds. Note also, in Figure IV-5, that the disk loading for both single and tandem rotor helicopters *decreases* as radius of action increases, for a constant gross weight; conversely, the disk loading *increases* as gross weight increases, for a constant radius of action. This trend is exhibited for all helicopter types investigated, as may be seen from the charts in Appendix F. It may be explained primarily by the fact that 1) Percentage fuel weight is greater for higher disk loadings, whereas percentage rotor system weight is lower, and 2) The percentage rotor system weight increases exponentially with gross weight for a given disk loading. Hence, at high radius of action where fuel weight is a significant item, minimum gross weight is achieved by reducing the disk loading. However, at higher payloads and correspondingly higher gross weights, the rotor system weight becomes rapidly predominant, and this is alleviated by higher disk loadings. This predominance of rotor system weight at higher gross weights is a primary reason why tandem rotor helicopters can be built lighter than single rotor helicopters above a certain size.

In addition to the information which appears directly on these charts, additional data may be readily calculated. Rotor diameter may be directly calculated from disk loading and gross weight; useful load is the sum of the payload plus fuel weight plus a constant 600 pound crew weight; and empty weight is simply the gross weight less the useful load.





IV-5 EXAMPLE DESIGN CHARACTERISTICS CHARTS FOR SINGLE AND TANDEM ROTOR HELICOPTERS WITH GEARED GAS TURBINE ENGINES

Chapter V
HELICOPTER OPERATIONAL FACTORS

A. Relationship of Operational Factors to the Measure of Effectiveness

For a given helicopter design configuration, the next step of the parametric analysis was to define a manner in which fleets of these helicopters would be operated.

In Chapter III, the various mission considerations were outlined and discussed, and in Chapter IV, the considerations leading to the selection of helicopter design configurations were covered.

It is the purpose of this chapter to summarize the assumed helicopter operational procedures and the various supplementary studies that were made to gain insight into the operational factors.

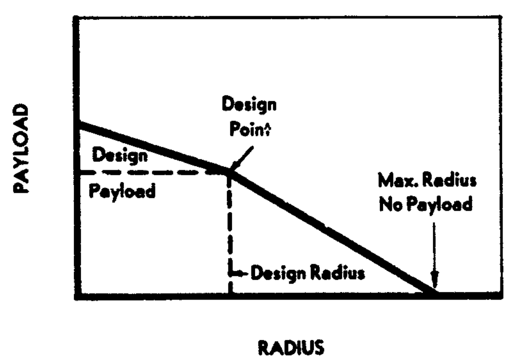
B. Gross Weight Operation

Initial concepts involved the study of cargo load factor on the various route structure segments, particularly in regard to the return trips from combat element supply dumps to the Army supply depot. Outbound supply requirements to the combat element dumps were readily established. In the absence of a specific return airlift requirement, it was assumed that the evacuation of casualties and equipment would be equal to the forward airlift. The number of aircraft required was determined by the outbound airlift and was inversely proportional to the design payload and the aircraft utilization. Although differences between outbound and inbound airlifts would result in a general reduction in absolute values of transport effectiveness, this could not influence the optimum decision.

C. Off-Design Point Payload

Since the helicopter design configurations were operated over various mission radii at full gross weight, they were not at all times operating at their design payload and radius. Under these off-design conditions, the aircraft operation was confined to the limits of the payload radius diagram for each particular configuration. A typical diagram is shown in Figure V-1.

Payload-Radius Relationship for a Given Design Point Helicopter



V-1 TYPICAL PAYLOAD RADIUS DIAGRAM

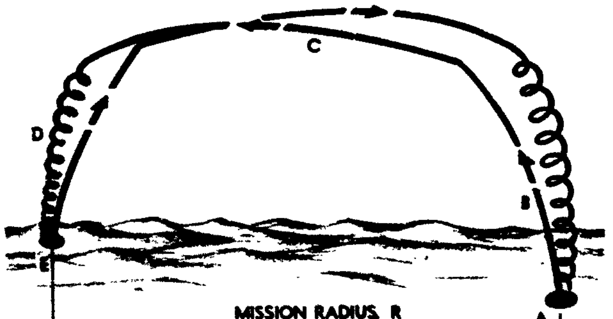
CHAPTER V — HELICOPTER OPERATIONAL FACTORS

Accordingly, where the aircraft was operated over a stage distance less than its design radius, additional payload could be carried. In such cases, the cargo carrying capability was limited by payload capacity only. Volumetric cargo limits were not considered. It was assumed that the aircraft could be loaded to its maximum allowable payload for the particular route segment involved from the many types of cargo available and required for airlift.

Adequate preliminary design layouts were made, however, to insure that certain sizes and weights of military equipment could be carried in the cargo compartment. This has been covered in detail in Chapter IV.

D. Mission Flight Plan Analysis

During the initial phases of this study, intuition as to the importance of certain operational factors led to the formulation of mission flight plans which adhered to typical maps laid out to depict the helicopter supply of a combat infantry division. These maps incorporated several route segments over variable distances and terrain altitudes. Analysis of daily operation required by the maps showed conclusively that the variation in stage length sequence and cruise altitude had negligible influence on the measure of effectiveness of any given helicopter fleet of one particular type and size, when compared to an "average" mission with a representative mean radius of action and cruise altitude. For this reason a standard mission, stipulating everything except radius, was selected as illustrated in Figure V-2, and all measures of effectiveness in this final report are based on this flight plan. It should be noted that this mission flight plan differs slightly from that given for transport helicopters in Military Specification MIL-C-5011A, which specifies a climb on course from sea level to 5000 ft., whereas the assumption has been made herein that an average climb



ASSUMPTIONS

Both home base and remote base ground elevation 4000 ft.

Standard NACA atmosphere, 45° F. at ground elevation, 41° F. at cruise altitude.

A. Start, warmup, * minutes at normal rated (maximum continuous) power.
B. Climb 1000 ft., on course, at best rate of climb, to cruise altitude of 5000 ft.
C. Cruise at 5000 ft. and best cruise power setting, to position directly above remote base.
D. Descend to remote base. No distance credit, no fuel used.
E. Land, stop engines, unload outbound cargo, load inbound cargo, start, warmup, and return to home base in accordance with above.

*Assumed time for start and warmup:

1. Reciprocating engines .... 5 minutes
2. Gas turbine engines .... 2 minutes
3. Pressure jets .... 2 minutes
4. Tip turbo jets .... 2 minutes
5. Ramjets .... ½ minute

V-2 MISSION FLIGHT PLAN

increment of 1000 ft. from 4000 to 5000 ft., is more representative of probable average operation. Certainly, terrain gradients of 5000 ft. in a very few miles do exist, but they are probably the geographical exception rather than the rule. In any case, the analysis indicated that only extreme deviations from this standard mission flight plan could change the optimum decision by a measurable amount.

E. Helicopter Fleet Utilization and Availability Studies

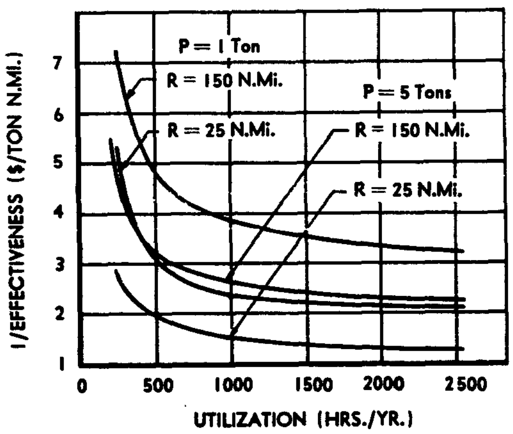
1. *Utilization Analysis*

A study was conducted for the purpose of examining the commercial and military helicopter utilizations exemplified by the past operation of presently available equipment.

It was found that the scheduled commercial helicopter air carrier operations within the United States were currently averaging 1000 to 1200 hours per year; and that this figure was limited only by scheduled frequency or flight hour requirement of the particular carrier. These same carriers felt that with presently available equipment, the present utilizations could be doubled if the airlift requirement would justify the 100% increase in flight time.

Utilization data used by military planning and procurement personnel for helicopter equipment similar to the equipment used by the above mentioned commercial carriers have been from 300 hours per year for the smaller types to 1200 hours per year for the large cargo types. The military utilization figures indicated are taken from Army Supply Bulletin SB 1-1. They apply to initial provisioning and mobilization rather than peace time conditions, and are used primarily for budgetary purposes. The relatively low utilization figures indicated for the smaller equipment are due, primarily, to the fact that the equipment is utilized in training and reconnaissance operations which exhibit by their very nature, low utilization.

Since no factual data existed on what the average helicopter utilization for a logistic airlift problem in a war situation might be, the effect of utilization variation for various payload-radius combinations on the military cost per ton nautical mile was determined and is presented in Figure V-3.



V-3 COST/TON N.MI. VS. UTILIZATION FOR VARIOUS DESIGN PAYLOAD RANGE COMBINATIONS

- Single rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude

The curves of Figure V-3 provided considerable insight into the aircraft utilization effect and, as might be intuitively deduced, minimum cost per unit of work capacity is realized as the aircraft utilization is increased.

It should be emphasized here, that since a sizable portion of the invested cost in a helicopter transport system varies inversely with

CHAPTER V — HELICOPTER OPERATIONAL FACTORS

the aircraft's utilization, a concerted effort be directed towards realization of maximum fleet utilization. Increases in fleet utilization have a direct effect in reducing the number of aircraft required to meet given airlift requirements and in lowering ton mile cost and therefore, in reducing military budget sizes. More military transport potential and capacity together with lower total costs are therefore realized with increases in transport fleet utilizations.

2. *Aircraft Availability and Loading Time*

It must be stressed that aircraft utilization is not synonymous with aircraft availability unless the operation is confined to one in which the payload is carried externally where the loading and unloading time approaches zero. The total aircraft availability can be defined as:

Availability = Flight utilization + loading and unloading time

Study was devoted to the topic of loading time and a considerable amount of information was gathered from the scheduled commercial fixed-wing carriers in this country, as well as from the Stanford Research Institute, Menlo Park, California. The latter has considerable experience in airlift studies, in which loading problems were of maximum importance. On the basis of the data collected, which are representative not only of helicopter operation, but also of the loading characteristics of many large transport airplanes, loading time was found to be a linear function of payload with a rate of approximately 4 minutes per ton. In the opinion of personnel concerned with the problems of loading large transport aircraft, the adoption of mechanized loading and unloading equipment and self-loading vehicles suitable for use in rough terrain, could ultimately reduce the loading time to 2.5 or 3 minutes per ton.

Since the time spent in loading and unloading was determinable as $K_1 P$ where

K_1 = Loading rate in hours per ton of payload

(Assumed equal to .133 for loading and unloading)

and P = Payload per aircraft per trip in tons

and would therefore vary with design configuration, it was necessary to fix either the aircraft utilization or availability. If the utilization were held constant for all design configurations, a variation in aircraft availability would follow. On the other hand if the availability were held constant, a variation in aircraft utilization would be enforced. The practice adhered to in this research was to fix the aircraft availability and allow a variation in aircraft utilization.

3. *Effect of Mission Concept on the Treatment of Availability and Utilization*

The consideration of any logistic transport problem enforces the following identity which is derived in detail in Appendix A.

$$U/A \left(1 + K_1 P \frac{V_B}{R}\right) = 1$$

Where U = Daily aircraft utilization or flight time in hours

A = Daily aircraft availability in hours

$K_1 P \frac{V_B}{R}$ = Ratio of loading time to flight time

This equation enforces the necessity for a variation in U/A with payload when the load is carried internally. The question of whether U or A should remain constant was determined as follows:

If the military logistic transport mission is considered as a routine supply problem wherein the total hours of operation per day are not limited and only a fixed airlift tonnage is required, the concept of a fixed utilization is reasonable.

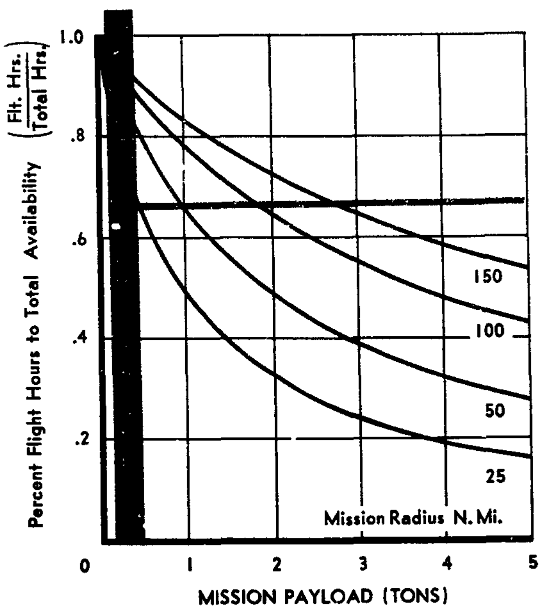
If, on the other hand, the mission is assumed to be one of extreme urgency at a particular time when not only the airlift tonnage requirement must be met, but there is also a limit on the time period within a day in which the mission must be accomplished, the concept of a fixed aircraft availability becomes more realistic.

In the opinion of the study group, the latter concept more closely defines the probable future use of transport helicopter fleets and a fixed aircraft availability was assumed. It must be pointed out however, that the value for this availability has no effect on the selection of the optimum system.

The effect of fixed availability on the optimum selection is to lower the optimum payload from that indicated as best under an assumption of fixed aircraft utilization. In other words, a smaller payload aircraft can spend more of its available hours in flight and can therefore deliver more payload per operating hour than a larger vehicle. This is shown clearly in Figure V-4.

It can be seen from Figure V-4 that as payload for a given trip is increased, the flight hour percentage of the total available hours decreases. This merely indicates that larger aircraft, requiring more time for loading and unloading, will suffer slightly in effectiveness, or cost per ton-nautical mile, since their aircraft utilization will be lower than for smaller vehicles. This is consistent with a fixed aircraft availability or a mission in which minimum time for accomplishment is required.

The second scale of Figure V-4 indicates availability in hours to a maximum of 5 hours per day which is the availability per day assumed for the results shown in Chapters VII,



V-4 EFFECT OF MISSION PAYLOAD AND RADIUS ON FLIGHT TIME AS A PERCENTAGE OF A FIXED AVAILABILITY

VIII and IX of this report. Under this assumption, it may be seen that the smaller aircraft operating over larger mission radii realize flight utilizations in excess of 1200 hours per year.

In calculating flight hour costs within the measure of effectiveness, therefore, utilization was determined as a function of payload and radius.

F. Helicopter Attrition

Combat attrition was not considered in this study. Normally consideration of this factor will materially affect the optimum system selection; however, in the case of the logistic support of Army combat elements, the opinion, requested from and advanced by U. S. Army personnel, was that no consideration be given to combat attrition, as the aircraft would seldom be operated in close proximity to actual battle areas. The most forward point of

CHAPTER V — HELICOPTER OPERATIONAL FACTORS

operations was assumed to be the regimental supply dump or the equivalent.

Military opinion, however, did indicate that operational attrition should be incorporated within the scope of the study. Military helicopter operations to date have suffered considerably from aircraft losses due to causes other than enemy action.

Most operational losses of transport helicopters can be related to two basic criteria:

1. Pilot technique and training
2. Probability of forced autorotational landing.

In the case of the latter, the two factors having predominating influences are probably the number of powerplants and the various component reliabilities.

In comparing past helicopter operations with those envisioned in this study, two major differences in operational attrition factors are of significance:

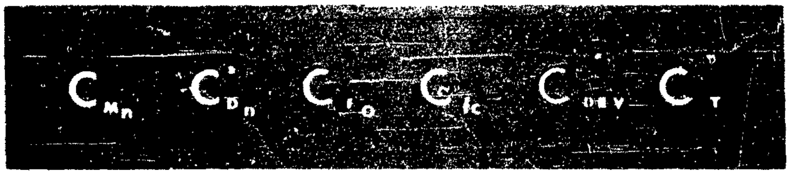
1. In operations to date, all of the aircraft have been single engine types, whereas this research has considered only multi-engine types.
2. In cruise flight average stresses in mechanical components of the helicopters in this study are lower than in currently operational military helicopters.

In past military helicopter operations, the aircraft involved have been designed with relatively high power loading with the result that in cruise operations there are many components which operate at high percentages of their design capabilities. This has brought about low overhaul periods and a general susceptibility to failure between scheduled overhaul periods with a net result of poor in-flight reliability.

The aircraft considered in this study, having lower design power loadings, cruise at relatively low percentages of their design capability which would result in increased flight reliability.

These aircraft, being multi-engined in addition to having lower percentage power requirements in flight than those which have been in operation to date, would be expected to incur lower overall operational losses.

The only military operational attrition data made available to the study group was not in detail. It indicated only that past Army helicopter operations exhibited an attrition rate of 2% per month and these losses could not be related to design parameters. Although it appeared that a reduction in operational attrition rate would seem reasonable, the 2% factor was conservatively used in this study. This would have no comparative effect on the optimum selections.



Chapter VI
TRANSPORT SYSTEM COSTING

One of the major study requirements dictated by the measure of effectiveness was the investigation of the various direct and indirect cost trends. Very little cost data pertinent to the subject of helicopter transport operation was available in any published literature, and it was necessary to conduct a broad survey of the major cost areas.

A. Development of Cost Criteria

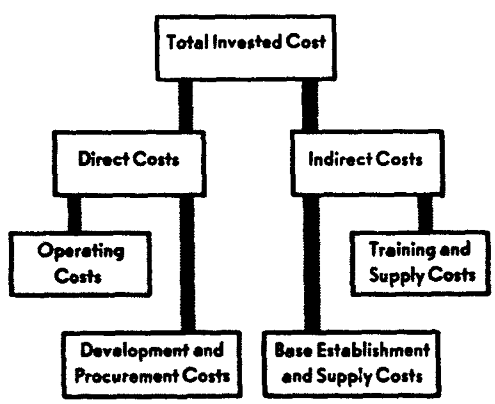
Many definitions can be given for the total expended cost of a military helicopter transport system, and more than one break-down can be used in separating direct from indirect cost expenditures. However, in order to define the cost problem so that a logical and systematic approach could be used in gathering and analyzing cost data, the break-down, which is shown as Figure VI-1, was initially established.

Direct costs were defined as those which were directly related to helicopter design and which were functions of its design variables. These included operating costs and procurement and development costs. The indirect costs were considered as those indirectly related to the helicopter design and which were primarily functions of helicopter and fleet size. These included such items as training and supply costs and base establishment and support cost. Within the operating cost area, the items of crew cost, fuel and oil cost, maintenance cost and depreciation cost were defined and investigated.

From previous analyses it was found that all of the above mentioned direct costs could be written as functions of the basic helicopter design parameters in such a way that they would lend themselves to a parametric analysis where their influence on design parameters could be studied and measured.

1. *General Costing Assumptions*

The costs of helicopter shipment to and supply support in any operational theatre has not been investigated or considered. It has been assumed that helicopter fleets are available in the proper areas at the proper time. This assumption would not invalidate the optimum decision, particularly since no attempt is made to compare the helicopter transport system with any other type of airlift or surface transportation where equipment shipment cost to any operational theatre might vary considerably from one type of transport system to another. Were this cost considered, the unit value



VI-1 INITIAL COST BREAKDOWN

of effectiveness for a given helicopter airlift system would decrease; but the same effect would be observed for all other systems.

The total costing, then, applies to those costs which are primarily attributable to helicopter transport establishment and operation and which reflect the supply of Army combat elements.

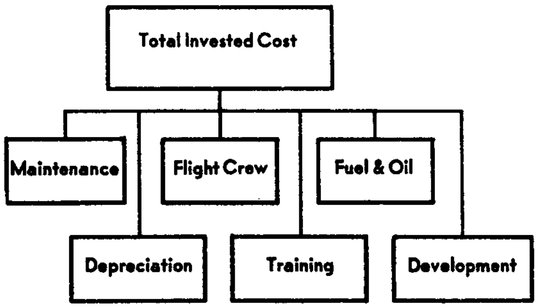
2. *Effect of Fixed Size of the Army*

During the cost survey, it became evident that the cost criterion should be based on the consideration of a fixed size of the Army. In this respect certain cost items would be present in one form or another, regardless of whether or not helicopters were utilized in air transport. Therefore it appeared that certain portions of the supply and base establishment cost need not be considered, since they could not be attributed to helicopter operation exclusively.

On the advice, therefore, of U. S. Army personnel, the consideration of training costs other than those specifically related to helicopter operation and maintenance training, together with base establishment and supply, were dropped. This was consistent with the assumption of a fixed size of the Army, which meant that preliminary and basic training programs, required for all military personnel, could not be attributed to the helicopter problem specifically. In addition, the problem of supply and support of these personnel could not be considered as attributable to helicopter operation; and lastly, the establishment of a helicopter base and the indirect supply of the base could not be attributed to helicopters alone, since, if helicopters were not employed, some other transport scheme would be; and base establishment and supply cost of a different nature, but of essentially the same general magnitude, would still be present.

3. *Final Elements of Cost*

Finally, then, the elements of cost which were considered to be attributable to helicopter utilization in military transport systems were as shown in Figure VI-2



VI-2 FINAL COST ELEMENTS

B. Helicopter Maintenance Cost Studies

The subject of helicopter maintenance costs presented a problem, since little information was available in the literature, and since very little insight existed in the industry as to the qualitative measurement of helicopter costs or their functional relationships.

It was necessary therefore, to collect as much maintenance cost data as possible from responsible personnel, representing a large number of helicopter operators.

At the present state of development and use of the military transport helicopter, available Army maintenance cost data appeared somewhat scattered as to source, indefinite in breakdown, and, in general, not of sufficient detail or consistency to be used in predicting trends or in estimating unit flight hour costs. Military cost data allowed an estimation of total cost factors, but it was necessary to utilize commercial operating cost information in establishing the trends of cost versus the pertinent variables. By using the trends based on commercial data, together with the total cost ratios found to exist between commercial total cost and military total cost, a reasonable estimate of the total maintenance cost was obtained for any given design configuration.

1. *Maintenance Cost Breakdown*

The maintenance cost study was divided into the consideration of five basic helicopter component groups as follows:

1.) Rotor Systems
2) Transmission and Drive Systems
3) Airframe
4) Engines
5) Other
(Radio and Instruments)

Sufficient detail in data was obtained so that total maintenance cost could be broken down into these five component groups. It was found that these component group maintenance costs exhibited a linear trend versus component group weight, as shown in Figure VI-3.

In addition, costs for mechanical power transmitting components were found to be closely related to the percentage of normal rated power required in cruise operation of the helicopter. This, in turn, is a function of equivalent parasite drag area per lb. of gross weight, design disk loading, cruise speed, and design power loading. A presentation of actual maintenance cost trends can be found in Appendix G.

Referring to Figure VI-3, it may be seen that the maintenance cost of any component group may be expressed as follows:

$$C_{M_n} = K_{1\,M_n} + K_{2\,M_n} \; R_n W$$

where

R_n = the component group weight to gross weight ratio

and

W = helicopter design gross weight.

The constant $K_{2\,M_n}$ is a function of the percent normal rated power required for cruise, and this must be determined by the technique presented in Appendix G. When a total maintenance cost for all the component groups is required, the equation takes on the form of

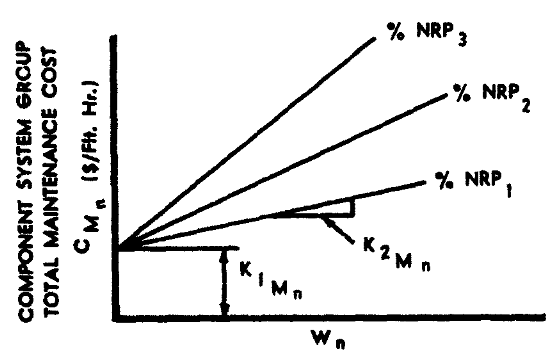
$$C_{M_T} = \sum_{0}^{n} \left[K_{1\,M_n} + K_{2\,M_n} R_n W \right]$$

Based on the data collected, however, this refers to a commercial flight hour cost, and furthermore, represents 1954 costs.

2. *Military Cost Level Correction—* K_{C_M}

As mentioned previously, the available data on military maintenance costs gave insight only to the total cost, and when this was compared with the commercial cost total, a factor of 2.5 was indicated, representing the ratio of military cost to commercial cost.

The factor does not consider the indirect cost of the support of the many maintenance personnel whose direct labor make up the labor cost portion of the total maintenance cost. It does not, therefore, include the cost of feeding, clothing and housing helicopter maintenance personnel. These factors are very often



VI-3 TYPICAL COMPONENT COST FUNCTION

included, but under the assumption of the fixed size of the Army, cannot be specifically attributed to helicopter operation.

Intuitively, the factor of 2.5 might well be expected, since the over-all complexity of military supply support systems are, of necessity, considerably more complicated than those found in similar commercial operations.

3. *Price Index Level Correction* — K_{PI}

Since the maintenance cost data collected was representative, on an average, of 1953 data, and since other costs collected for this study were representative of other time periods, a price index correction was developed which would allow the adjustment of all costs to the mean time period of the study, mid-1958. All costs used in the study, then, are based on the extrapolated value for mid-1958 dollar.

Considering the price index factor, together with the military cost level correction, the equation for total maintenance cost takes on the form shown below.

$$C_{M_T} = K_{PI}\; K_{C_M} \sum_{0}^{n} \left[K_{1_{M_n}} + K_{2_{M_n}}\; R_n W \right]$$

For the many helicopters considered within the scope of this research, the ratio R_n was determined analytically as well as the value of minimum gross weight for the particular payload-range combination being considered. This, together with the analytical determination of those factors influencing percent normal rated power used in cruise operation, allowed the calculation of total maintenance cost for any helicopter design configuration within the scope of the study.

4. *Scope of the Maintenance Cost Data*

The statistical cost information gathered from the helicopter operators contacted during the cost survey was based on those aircraft which are currently operational. This meant that the component group weight range to which the data applied was relatively small in comparison to the weight ranges necessitated by the design configurations involved. For this reason, an appreciable extrapolation of the statistical cost data was necessary to predict maintenance cost of helicopters with empty weights approaching 70,000 pounds. The linearity of this extrapolation was based on the correlation of the collected data, together with the history provided by the scheduled fixed wing carriers whose experiences over the years have indicated a linear relationship of maintenance cost with component system weight.

Actual tandem helicopter maintenance costs are not reflected in the statistical cost information. To date, there have been no tandem configurations used in commercial helicopter operations, and since the maintenance cost trends were based on commercial operators' statistical data they do not indicate whether a difference in unit flight-hour cost per pound of component weight would exist between single rotor and tandem rotor configurations.

Military cost data[1] was examined in an effort to settle this problem but was found to be inconclusive since the data for only one tandem rotor type was represented.

Fixed wing and helicopter air carrier personnel were also questioned regarding their intuition on the problem of possible difference between tandem and single rotor unit costs per pound of component weight. Most replies evidenced a general feeling that the unit cost per pound of component weight for groups

[1] *Analysis of Army Aircraft Operating and Maintenance Costs* Project 9-72-02-001, July, 1955—Army Aviation Div. TRADCOM, Fort Eustis, Virginia.

having a multiple installation of engines, rotors or transmissions, would be higher. The following intuitive analysis was offered by one of the air carrier personnel contacted and was substantiated by others:

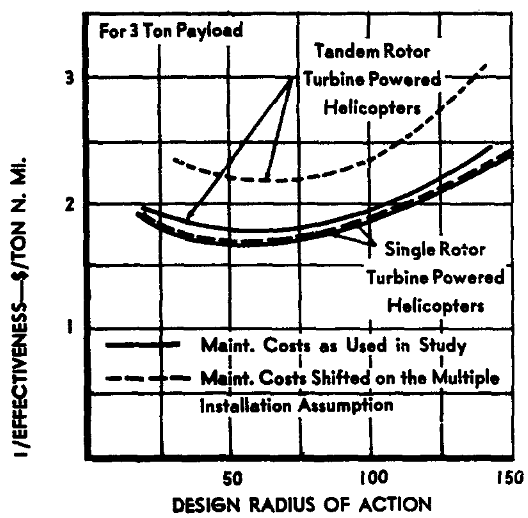
a) Material costs, whether for single or multiple installations of engines, rotors and transmissions, would be directly proportional to the total component weight and would therefore be adequately predicted by the trends of material cost vs. component weight developed from single rotor configuration statistical data.

b) Labor costs would be directly proportional to the number of installations for any one component group, such that doubling the number of installations would be accompanied by an 80 % increase in labor cost.

For equal component group weight in a tandem and single rotor machine then, the material costs would be identical but labor costs for transmissions and rotors in the tandem would be 1.8 times the labor costs for these component groups in the single rotor machine. This would also mean an 80% increase in engine labor cost for the single rotor configurations having twin engines rather than a single powerplant. This 80% factor however, does not represent a statistical survey of data but merely reflects the intuition of personnel contacted.

Since the survey resulted in no statistical justification of an increase in tandem rotor configuration unit maintenance costs over the single rotor configuration unit costs, they were assumed to be identical in calculating the results presented in chapters VII, VIII and IX. It was felt by the study group that pure intuition, regardless of its source or support, should not be used in the optimum systems evaluation.

However, in order to estimate the effect of the multiple installation maintenance cost factor, should future operations prove its existence, a selected group of helicopter configurations were processed through the measure of effectiveness with their maintenance costs altered to include the factor. The trends in military cost per ton nautical mile for this consideration are compared to those without the factor included, in Figure VI-4.



VI-4 EFFECT OF MULTIPLE INSTALLATION MAINTENANCE COST INCREASE ON TOTAL MILITARY COST PER TON-N.MI.

As may be seen from Figure VI-4, the effect on engine maintenance cost alone for the single rotor configuration was negligible. The combined effect however, of the multiple installation maintenance cost factor on engines, rotor systems and transmissions and mechanical drives systems of tandem configurations, accounted for an average increase in total cost per ton nautical mile of about 25%. If the multiple installation maintenance cost increase of 80% on labor cost were completely accurate and statistically justifiable, the data presented in Figure VI-4 would have eliminated any further consideration of tandem rotor types for the logistic support of army combat elements.

C. Helicopter Component Depreciation Cost Analysis

In devising a suitable method for determining depreciation of first and spare parts cost write-off, a number of commercial operators in the United States were contacted, and it was found that the differences between methods used were related to the particular operator's organizational structure, financial situation and tax problems.

Enough basic information was obtained however, so that the results of the depreciation cost study could be based on a rational approach. Thus, the technique was applicable to any military helicopter operation. The cost data was obtained from several manufacturers, and comprised both total aircraft first cost and spares cost. The study was confined to rotary wing manufacturers for the establishment of present price levels for various component types, but fixed-wing manufacturers' data was used in the determination of some price trends with aircraft component size or weight.

1. *Component System Break-down*

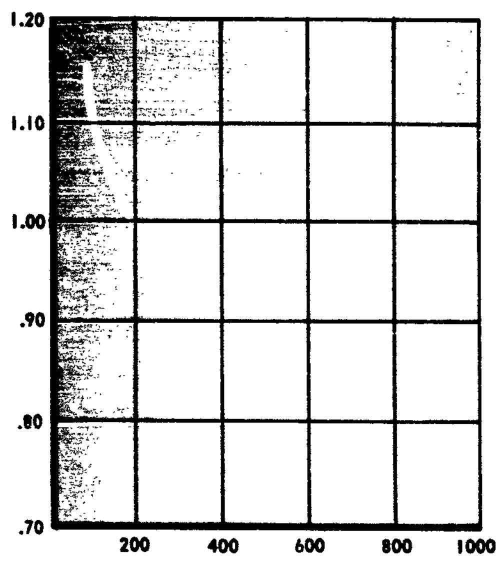
As in the maintenance cost analysis, the helicopter was broken down into the same five component groups and the per pound unit costs were established, assuming as a base the production run of 200 aircraft. These unit costs are shown in Figure VI-5.

2. *Production Quantity Correction*

In order to allow the adjustment of the unit cost for production quantity, the production quantity unit cost correction factor was developed and is presented in Figure VI-6.

	Component System Group	Dollars/Lb. ($K_p = 1$)
	n	C_{U_n}
	Airframe	\$34.50
	Rotor System	\$28.50
	Transmission and Mech. Drives System	\$43.10
Engines	Reciprocating	\$20.00
Engines	Shaft Turbines	\$44.00
Engines	Tip Turbines	\$50.00
Engines	Ramjets	\$22.00
	Other (Radio & Instruments)	\$17.25

VI-5 COMPONENT GROUP UNIT FIRST COSTS



VI-6 PRODUCTION QUANTITY CORRECTION FACTOR

As may be noted from the measure of effectiveness, no term representing the number of aircraft required appears directly. What is indicated by the measurement of effectiveness is the total number of helicopter flight hours for any particular configuration to meet the requirement. When this has been determined and a suitable average aircraft utilization figure is employed, the number of aircraft required becomes known. It may be seen, therefore, that for the number of production units to be accurately defined, K_P must be obtained through an iterative process. However, K_P, as presented herein, is based on a single source of production. If a second source of production is considered when the number of aircraft required is in excess of the value used in the initial determination of K_P, the entire situation changes; and the K_P value corresponding to the second source of production must reflect the second source production quantity.

In the processing of cost data however, for all of the design configurations within the scope of the study, it was found that percentage-wise, the effect of successive approximations on K_P was negligible. The effect, therefore, on the measure of effectiveness or the selection of optimum systems was negligible. The actual practice was to employ an average value of K_P for a production quantity of 200 aircraft for all design considerations.

The depreciation cost write-off for a basic component system was expressed as follows:

$$C_{D_n} = \frac{K_R \, C_{U_n} \, W_n}{D_P U}$$

where C_{U_n} = component group unit first cost (dollars per lb.)

W_n = component group weight (lbs.)

K_R = residual constant (1 minus the fractional residual value at the end of the depreciation period)

D_P = Depreciation period (years)

U = yearly average aircraft utilization (hours per year)

Upon advice from Army sources, a depreciation period of five years was considered as typical of a military write-off time. This period was assumed for all component groups.

In view of the fact that a good number of obsolete military aircraft are sold on the Government surplus market, a conservative residual value of 5 percent of the initial cost was assumed for all component groups. This gave a residual constant value of K_R = .95.

The preceding equation can be rewritten in the form:

$$C_{D_n} = \frac{K_R \; C_{U_n} \; R_n \; W}{D_P U}$$

where the term $R_n W$ is the component group weight in terms of the component group weight to gross weight ratio, as in the maintenance cost analysis.

3. *Spares Cost Write-Off*

The write-off cost of all spares must also be included in the depreciation cost before the total depreciation costs have been covered. The number of spare parts required to support any helicopter operation is a function of the following three variables:

1) Aircraft utilization
2) Scheduled overhaul period
3) Time factor for a component to be shipped to an overhaul base, overhauled, and returned to stock.

For preliminary budgetary purposes, U. S. Army sources advised the following formula for the determination of spare parts requirement:

$$\delta_n = \frac{t_{Tn} \quad U/12}{OHP_n}$$

δ_n = average number of component group spares per aircraft

t_{Tn} = average component group turn-around time (months)

U = average aircraft utilization (flight hours per year)

OHP_n = average overhaul period of parts within a particular component group (flight hours)

An average three-month turn-around time was assumed, and when the spares cost write-off was combined with the first cost write-off, the equation for total depreciation cost for any component group took on the form:

$$C_{Dn} = \frac{.19\, K_P\, K_{PI}}{U} \times \left[C_{un}\, R_n\, W \left(1 + \frac{U}{4 \times OHP_n} \right) \right]$$

It may be seen that the above equation accounts for the production quantity correction, as well as the price index adjustment mentioned in the maintenance cost discussion.

4. *Overhaul Period Estimation*

Based on techniques developed in the maintenance cost study and on the advice of both commercial and military operators, the following overhaul periods were used for the purposes of obtaining estimates on required spares support:

1) Engines, determined from techniques developed in maintenance cost study
2) Transmission and mechanical drives, determined from techniques developed in maintenance cost study
3) Rotor system — 1000 hours
4) Airframe — 8000 hours
5) Other — 1000 hours

Some of these values were selected arbitrarily, but no effect on the optimum transport system selection could be attributed to this choice, since all aircraft within the scope of the study were treated on exactly the same basis.

The final equation presenting the complete first and spares cost depreciation for an entire helicopter is given by the following relationship:

$$C_{D_T} = \frac{.19\, K_P\, K_{PI}}{U} \times \sum_{1}^{n} \left[C_{un}\, R_n\, W \left(1 + \frac{U}{4 \times OHP_n} \right) \right]$$

where the summation from 1 to n allows the compilation of the first and spares cost write-off on all component groups.

The data needed in the above equation was determined from the design analysis techniques for any configuration within the scope of the study.

D. Fuel and Oil Costs

A study of fuel and oil costs, based on bulk sales to the U. S., within the continental limits, was carried out for the various power plant types.

For a particular mission, stage distance, or route segment, the total flight hour fuel and oil costs were expressed by the following equation:

$$C_{f_0} = K_{PI} \, K_{f0} \, \frac{r_F}{t_j} \, W$$

where

K_{PI} = the price index adjustment factor mentioned previously under maintenance and depreciation costs to bring the collected price information to the level of the mid-point of the study.

r_F = fuel weight to gross weight ratio to meet the range requirement of any j th segment within a given route structure.

K_{f0} = equivalent cost per lb. of both fuel and oil.

t_j = total block time to cover the j th segment of any given route structure.

The fuel and oil cost can be converted to a slightly different form, making use of block speed; but for purposes of detailed analysis of fuel and oil costs, the equation above is more useful when the operation consists of many route segments.

E. Crew Costs

The flight crews on all helicopters considered within the scope of the study were assumed to consist of:

W/O Pilot
W/O Co-pilot
W/O Flight Engineer

Using the average annual pay scales for this grade, the expression for crew cost was simply

$$C_{FC} = \frac{K_{PI}}{U_C} \, C_{C_A}$$

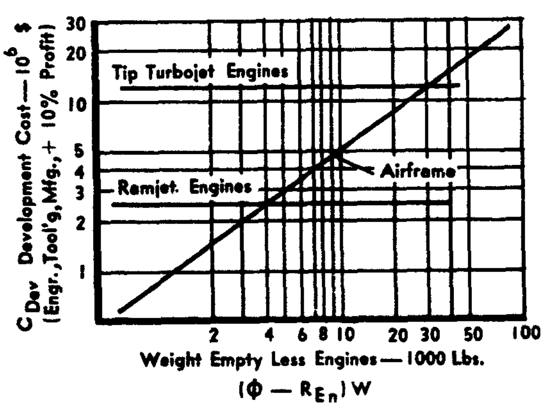
where C_{C_A} is the total annual crew cost and U_C is the assumed average flight crew utilization (yearly).

Crew utilizations of 1000 hours per year were assumed for all helicopters within the scope of the study. This amounts to less than 3 flight hours per day, and is believed to be realistic for an actual military operation. It is further justified by the experience of the Military Air Transport Service in operation of large fixed-wing transport aircraft, which has experienced the same value. Using these figures, a crew cost of $15.37 per flight hour was considered for all configurations.

F. Development Cost

The development cost study, which utilized available helicopter development costs as a basis for establishing a level and fixed wing cost data for establishing the trend with weight, was made in an effort to obtain quantitative information which could be related to the aircraft's design parameters.

Shown in Figure VI-7 is a plot of total development costs of airframes, including production engineering, tooling, manufacturing



VI-7 DEVELOPMENT COSTS

and ten percent profit, with the engine development costs for ramjets and tip turbojets also indicated.

Engine manufacturers' data indicates that the development costs for tip turbojet or ramjet engine power plants of the size applicable to configurations within the scope of this study are essentially constant with engine weight. The airframe development cost data is plotted versus the basic weight empty less engine weight.

The equation representing the airframe development costs can be written in the form:

$$C_{dev} = \frac{K_D K_{PI}}{P_W U N_S} \left(\phi - R_{EN}\right)^{\epsilon} W^{\epsilon}$$

This equation puts the development cost on an average flight hour basis, and, as may be seen, the price index correction has again been applied. Development cost data, presented in the curve, is indicative of 1954 prices, and, therefore, was adjusted to the midpoint of the study.

P_W = write-off period (assumed five years)

U = aircraft average utilization (flight hours per year

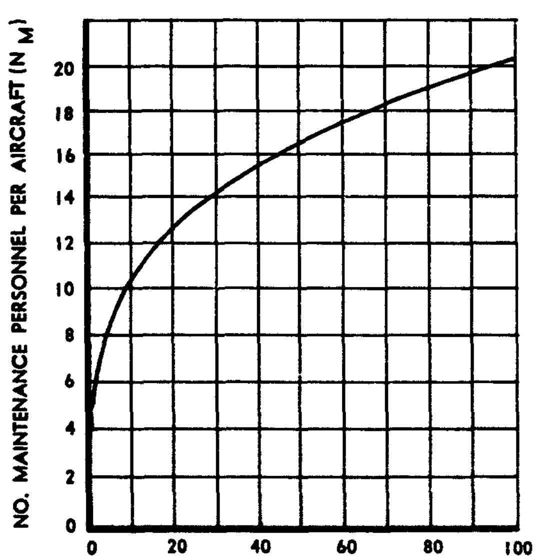
N_S = number of ships procured.

The same procedure as mentioned in the depreciation cost analysis discussion with regard to the production quantity adjustment factor was applied in selecting the proper value of N_S

G. Training Costs

Military training cost data indicated helicopter pilot training cost to be $36,000, and helicopter mechanic training costs of $3900. These costs include field and organizational maintenance training, student pay, fuels, instructors, direct cost of supervisors, training aids, and a proportionate amount of the indirect costs chargeable to the training program. The flight crew was assumed to consist of a pilot, co-pilot and flight engineer for all helicopters considered. Flight engineer training costs were assumed to be twice that of mechanic training costs, in the absence of specific information for this category. This gave a total flight crew training cost of approximately $80,000.00.

For the calculation of mechanic training costs, the number of mechanics per aircraft was based on the curve of Figure VI-8 which was derived from commercial helicopter operators' data, and includes the total depot overhaul maintenance support as well as line and second echelon maintenance on all components.



VI-8 EFFECT OF HELICOPTER EMPTY WEIGHT ON TOTAL NUMBER OF MAINTENANCE PERSONNEL REQUIRED PER HELICOPTER

The equation used for predicting total training cost on a flight hour basis was:

$$C_T = \frac{K_{PI}}{UP_S}\left[C_{T_{FC}} + K_{C_M}\, C_{T_M}\, N_M\right]$$

Where

$C_{T_{FC}}$ = Flight crew training cost in dollars ($80,000.00)

K_{C_M} = Military cost correction factor (2.5)

C_{T_M} = Mechanic training cost ($3900.00)

N_M = Number of mechanics per aircraft

P_S = Average service period of the flight crew and maintenance personnel (years)

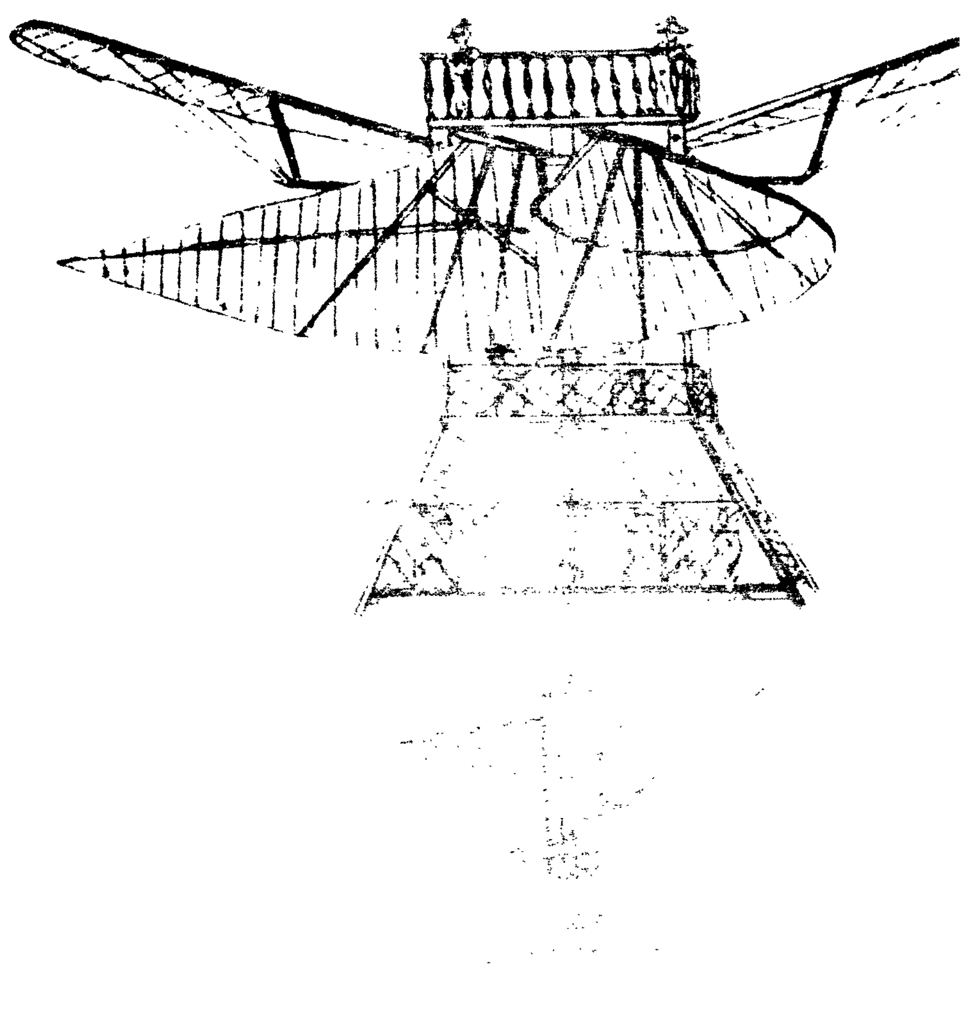
U = Aircraft utilization (hours/year)

K_{PI} = Price index correction factor

Although the factor K_{C_M} has been shown previously as a cost scale up factor, military maintenance and manpower statistics have indicated the justification of its application to labor hours and manpower as well. The factor K_{C_M} was therefore included to allow for the additional mechanics in training together with rear base and Zone of the Interior maintenance supply and support personnel.

For the purposes of this study, the service period was assumed as four years, which is in excess of the average now realized. Although the flight hour training cost is sensitive to the period of service, it must be emphasized that the training cost formed only a small portion of the total cost for all configurations.

RESULTS



Chapter VII

EVALUATION - 1956 to 1961

A. Data Processing

The basic results derived from the parametric evaluation of the helicopter configurations considered to be production possibilities in the time period 1956 to 1961 are presented in this chapter. This section of the chapter presents a brief outline of the steps involved in evaluating the cost per ton-nautical mile, I/E, for each of these configurations. The identical procedure was used in the evaluation of tip-powered helicopters considered as near-future possibilities for the time period 1960 to 1970.

The point of departure for the processing of the final data is represented by the *design characteristics charts* (Chapter IV and Appendix F). From these charts, the *minimum design gross weight* and corresponding *disk loading* and *fuel weight* were obtained for each combination of design radius-of-action and design payload.

The major task involved in processing the data, as may be seen by inspection of the *effectiveness equation* (Chapter II), was the calculation of the total flight-hour cost for each configuration. In addition to the primary *design characteristics* data, the total flight-hour cost computation involved the tabulation of: cruise speed, average rate-of-climb, fuel consumption rate in cruise, percent normal rated power setting in cruise, and the component group weight ratios of the rotors, airframe, engines, transmissions and drives, and "other" (radio and instruments). *Of these factors which influenced the total cost per flight-hour, the percent power setting in cruise and the component group weights were by far the most predominant, through their influence on maintenance cost, which was in all cases the largest single cost factor.*

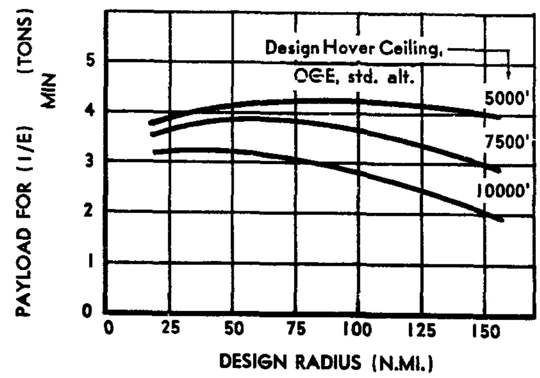
Consideration of the *flight hours* per *available hour* led to the selection of yearly helicopter utilizations depending upon the *loading time* (a function of design payload), as discussed in Chapter V. As noted therein, the total daily availability was assumed to be 5 hours, or 1825 hours yearly, and the utilization was computed as the difference between the available hours and the total time spent in loading and unloading payload. This utilization was then used as the basis in computing the total cost per flight-hour.

Variations of this technique were used to evaluate the effects of *off-design operation* (i.e. full gross weight operation, but at different payloads and radii-of-action than those for which the helicopter is designed), and the effects of *externally carried payload*. For these special analyses, the affected terms in the measure of effectiveness equation were modified as required. For the analysis of off-design operation, the modification involved 1) the computation of the incremental fuel weight, positive or negative as radius of action was increased or decreased from the "design point", and 2) the addition of this increment to the design payload. For increased radius of action, allowance was made for auxiliary fuel tank weight. Flight-hour cost for the off-design operations were assumed to remain unchanged, as analysis showed the attendant changes in fuel cost to be a negligible percentage of total cost. For the analysis of externally carried payload, the modification involved 1) the reduction of the block speed due to the higher drag, 2) the reduction of payload due to the higher fuel consumption rate and hence higher fuel weight for a given radius of action, and 3) the reduction of the flight-hour cost due to the elimination of time lost in loading and unloading payload, resulting in higher utilization.

B. Transport Effectiveness Trends, Basic Study, 1956-1961

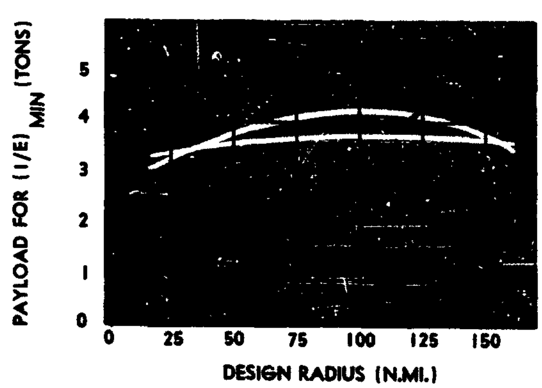
The basic study results showed, as expected, that there is a different *optimum* design payload (the optimum being that value which produces maximum effectiveness, or minimum cost per ton-nautical mile, 1/E) at each design radius of action. Figures VII-1 through VII-4 show this variation of optimum design payload with design radius of action, for the single rotor and tandem rotor helicopters, with reciprocating and geared gas turbine power plants. Curves are included in these charts for each of the three design hover ceilings considered. Figures VII-5 through VII-8 show the typical trends of 1/E versus design payload, for various design radii of action, and for a design hover ceiling of 5000 feet as an example. It is from these and similar charts for design hover ceilings of 7500 and 10000 feet that the optimum payloads of Figures VII-1 through VII-4 were obtained.

Some understanding of the reasons behind these trends may be had, when it is recognized that *increasing payload causes a decrease in cost per ton-n.mile only up to a certain point, beyond which the empty weight, fuel weight, and attendant costs increase more rapidly than the work capacity (ton-n.miles per hour).* Therefore, since the installation of higher



VII-1 PAYLOAD FOR MINIMUM COST/TON N.MI. VS. DESIGN RADIUS FOR VARIOUS HOVER CEILINGS

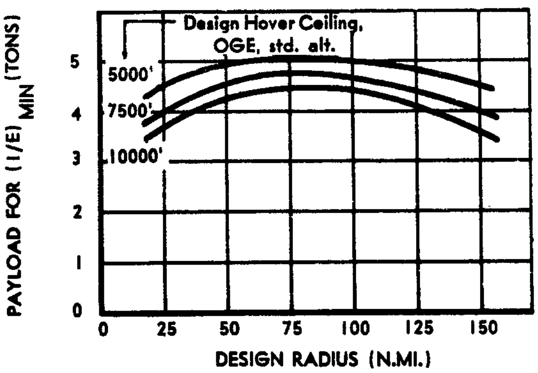
- Single rotor helicopter
- Twin reciprocating engines



VII-2 PAYLOAD FOR MINIMUM COST/TON N.MI. VS. DESIGN RADIUS FOR VARIOUS HOVER CEILINGS

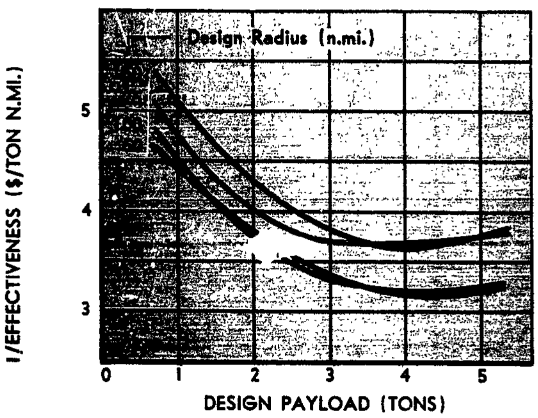
- Single rotor helicopter
- Twin geared turbines

CHAPTER VII — EVALUATION — 1956 TO 1961



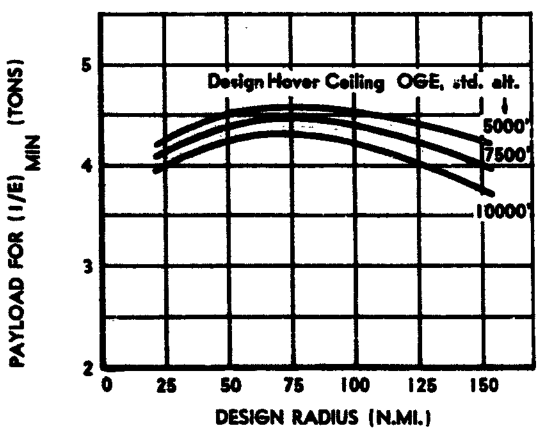
VII-3 PAYLOAD FOR MINIMUM COST/TON N.MI. VS. DESIGN RADIUS FOR VARIOUS HOVER CEILINGS

- Tandem rotor helicopter
- Twin reciprocating engines



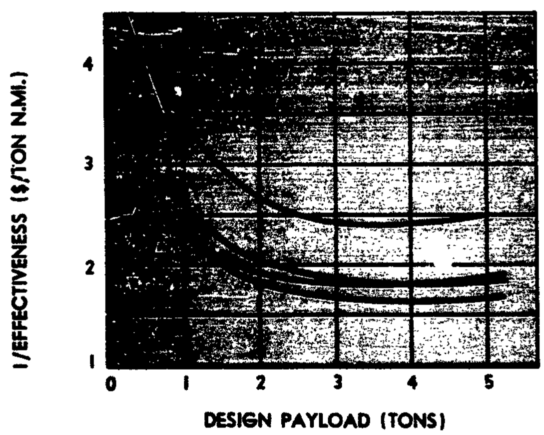
VII-5 COST/TON N.MI. VS. DESIGN PAYLOAD FOR VARIOUS DESIGN RADII

- Single rotor helicopters
- Twin reciprocating engines
- 5000' design hover ceiling, OGE, standard altitude



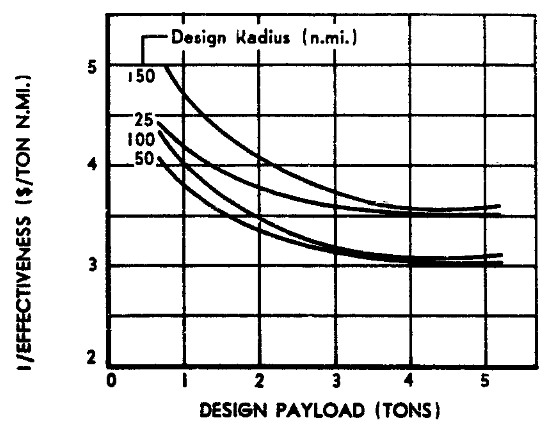
VII-4 PAYLOAD FOR MINIMUM COST/TON N.MI. VS. DESIGN RADIUS FOR VARIOUS HOVER CEILINGS

- Tandem rotor helicopter
- Twin geared turbines



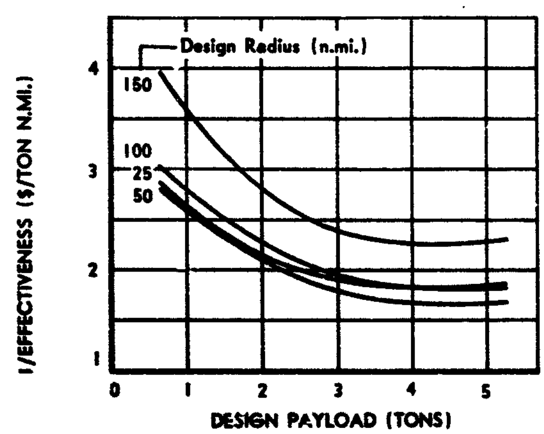
VII-6 COST/TON N.MI. VS. DESIGN PAYLOAD FOR VARIOUS DESIGN RADII

- Single rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



VII-7 COST/TON N.MI. VS. DESIGN PAYLOAD FOR VARIOUS DESIGN RADII

- Tandem rotor helicopters
- Twin reciprocating engines
- 5000' design hover ceiling, OGE, standard altitude



VII-8 COST/TON N.MI. VS. DESIGN PAYLOAD FOR VARIOUS DESIGN RADII

- Tandem rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude

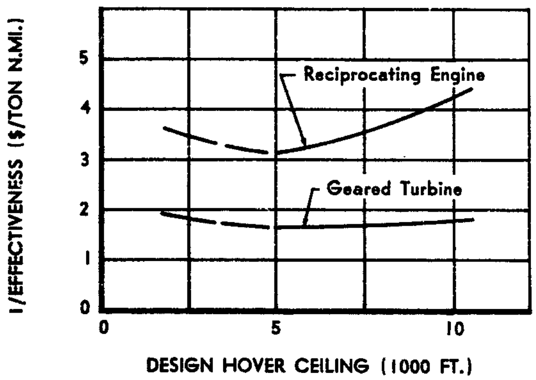
power to achieve higher hover ceilings in itself results in an *empty weight growth,* it effectively reduces the "point of diminishing returns", forcing the minimum 1/E to occur at a lower design payload. Similar effects may be noted when payload and hover ceiling are held constant and radius of action is increased. In this case the trends also indicate an *optimum* radius of action at which 1/E is minimized, beyond which weight and attendant cost increases are again predominant. Thus, it is apparent that the "state of the art" dictates a certain size bracket within which the best transport helicopter design, from the standpoint of minimum cost per ton-n. mile, will be located.

To summarize, the reasons for most of the trends in 1/E illustrated in this and the following chapters may be better understood if the following axiom is kept in mind: *All factors which cause a weight growth, such as increasing payload, hover ceiling, radius of action, or other performance improvement, will inevitably cause an increase in cost per ton-n. mile beyond some "point of diminishing returns" at which the weight growth and attendant cost become predominant.* This phenomenon is manifested to a greater or lesser extent in all design types, and the "point of diminishing returns" in terms of size, is dependent upon the relative predominance of each factor. It is emphasized that increases in power have a twofold effect. On the one hand, percent power in cruise is decreased, thereby tending to decrease the maintenance costs, and on the other hand, the necessary weight growth tends to increase all costs.

C. Effect of Hover Ceiling and Temperature

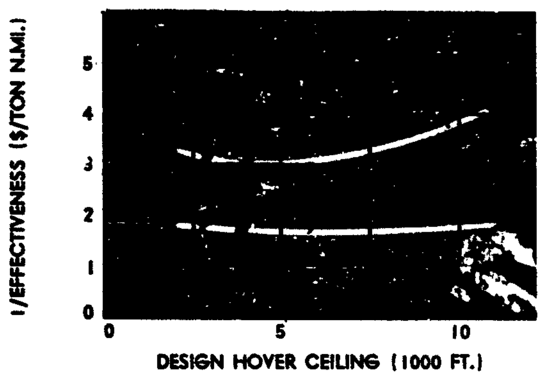
A general understanding of the effects of hover ceiling and operating temperature re-

CHAPTER VII — EVALUATION — 1956 TO 1961



VII-9 COST/TON N.MI. VS. DESIGN HOVER CEILING, O.G.E., STANDARD DAY

- Single rotor helicopters
- Cost per ton n.mi. based on optimum design payload and range for the hover ceiling



VII-10 COST/TON N.MI. VS. DESIGN HOVER CEILING, O.G.E., STANDARD DAY

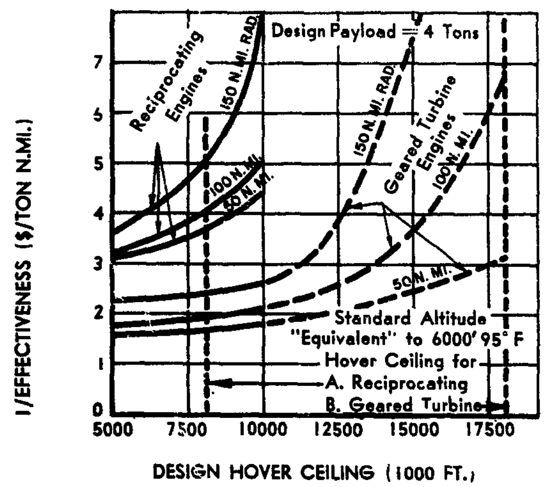
- Tandem rotor helicopters
- Cost per ton n.mi. based on optimum design payload and range for the hover ceiling

quirements on helicopter cost per ton-nautical mile may be had from the trends shown in Figures VII-9 through VII-12. These trends are shown only for the geared power plants of the 1956-1961 analysis; however, those for the geared turbine engines are qualitatively similar to those which may be expected for tip turbojets, pressure jets, and ramjets.

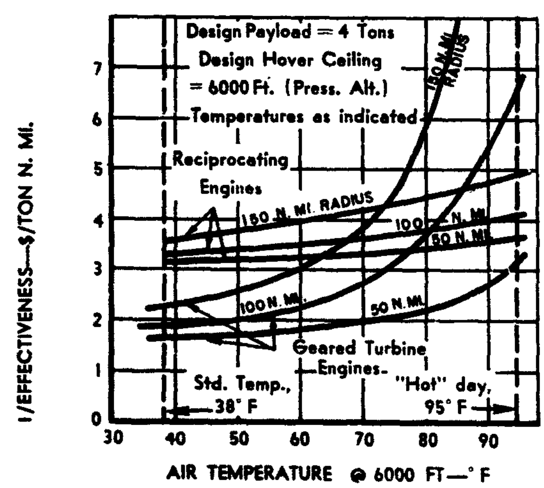
In Figures VII-9 and VII-10, curves of 1/E versus design hover ceiling, (in standard atmosphere, out-of-ground effect, using normal rated power), are shown for reciprocating and geared turbine power plants, and for single rotor and tandem rotor helicopters. These are *hybrid* curves, made up of the *minimum* 1/E values at different payloads and corresponding radii of action, from the curves in the preceding Section B. It may be noted that the geared gas turbine curves are nearly flat over the range of hover ceilings from 5000 to 10000 feet, the minimum 1/E occuring at or below 5000 feet. The reciprocating engine 1/E curves also minimize at or below 5000 feet, and show a more marked increase as hover ceiling requirements are increased. This may be explained by the fact that reciprocating engine weight and cost are considerably more predominant factors than geared turbine weight and cost. Below 5000 feet hover ceiling, both types were estimated to exhibit a slight increase in 1/E primarily due to higher cruise percent power settings for the lower power installed, resulting in higher maintenance costs. The reciprocating curves actually show a discontinuity, or break in slope, at 5000 feet. This discontinuity is a manifestation of the assumed 5000 ft. critical altitude of the supercharged reciprocating engines.

Figure VII-11 is a representative illustration of the penalty to be paid in higher cost per ton-nautical mile in return for higher hover performance, for a pre-selected design payload. The example is for single rotor helicopters with a design payload of 4 tons (reasonably close to the optimum payload at all radii of action, as may be seen from Figures VII-5 and VII-6). Both reciprocating and geared gas turbine configurations are shown on this chart, and the five different design radii of action are included as separate curves. Note that the penalizing effect is more pronounced for the higher radii of action, a trend due again to the *weight growth* phenomenon discussed in Section B. Note also that the standard altitude hover ceiling which would be equivalent to a ceiling of 6000 feet at 95°F is approximately 18000 feet for the geared gas turbines, and only about 8000 feet for the reciprocating engines. This large differential is a result of the fact, previously discussed, that geared gas turbines suffer an approximate 30% power loss under these "hot day" conditions, whereas the reciprocating engines suffer only about 5% power loss.

An example illustration of the effects of increasing the operating temperature requirements at a given hover ceiling is presented in Figure VII-12, which is simply a conversion of the curves shown in Figure VII-11 to a temperature scale, for a fixed hover ceiling of 6000 feet pressure altitude. Here the lower penalties to be paid with reciprocating engines as operating temperature requirements are increased are clearly demonstrated. However, the 1/E values for the geared gas turbines are so much lower than for the reciprocating engines at the *standard temperature* condition, (38°F), that the cross-over points at which the reciprocating engines would be competitive occur at rather high temperatures, from 74°F for 150 nautical miles, to temperatures in excess of 95°F for 50 nautical miles.



VII-11 TYPICAL VARIATION OF 1/E WITH DESIGN HOVER CEILING (O.G.E.), STD. DAY. COMPARISON OF RECIPROCATING AND GEARED TURBINE ENGINES. SINGLE ROTOR HELICOPTERS, DESIGN PAYLOAD = 4 TONS.



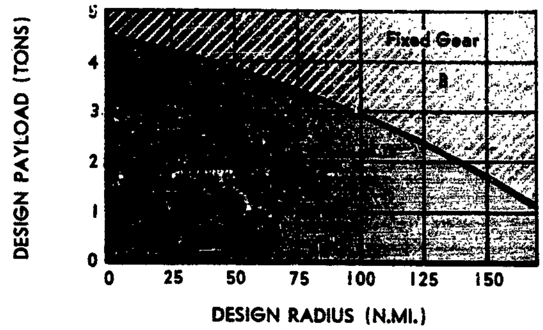
VII-12 EFFECT OF AIR TEMPERATURE ON 1/E FOR SPECIFIED HOVER CEILING OF 6000 FT. O.G.E. (PRESSURE ALTITUDE). COMPARISON OF RECIPROCATING AND GEARED TURBINE ENGINES. SINGLE ROTOR HELICOPTERS, DESIGN PAYLOAD = 4 TONS

D. Effect of Retractable Landing Gear

A special analysis of the effects of retractable landing gear installations was made using single rotor geared turbine powered helicopters only, since it is to be expected that the effects would be quite similar for the tandem rotor types. A constant 40% increase in landing gear weight over the weights for fixed landing gear was assumed. Aerodynamically, the cruise speeds were increased due to reduced drag only in a few cases where cruise speed with fixed gear had been less than the rotor-limited maximum speed (111 knots at 5000 feet). For all other cases the cruise speeds remained at 111 knots at 5000 feet, and the primary aerodynamic effect was the reduction of fuel consumption rate due to the lower drag. Development, depreciation, and maintenance costs for the retractable landing gear were assumed to be the same *per pound* as for fixed landing gear, hence the only cost increase was due to the *weight growth.*

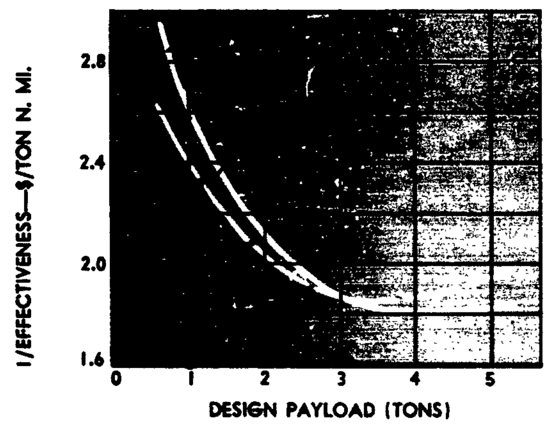
The important aspects of the results of these analyses are illustrated in Figures VII-13 and VII-14. The trends shown in Figure VII-13, which defines the design payload and design radius of action regime (A) within which retractable landing gear produces lower 1/E, and the regime (B) in which fixed landing gear produces lower 1/E, are yet another manifestation of the *weight growth* axiom previously stated. Because of the weight growth phenomenon, heavier retractable landing gear are competitive only at lower sizes and gross weights (corresponding to the regime of lower payload and radius of action, A), whereas lighter fixed landing gear produce lower values of 1/E at larger sizes and gross weights (corresponding to the regime of higher payload and radius of action, B).

Intuitively, the reverse of the trends shown



VII-13 DECISION CHART FOR LANDING GEAR TYPE

- Single rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



VII-14 EFFECT OF RETRACTABLE LANDING GEAR: COST/TON N.MI. VS. PAYLOAD

- Single rotor helicopters
- Twin geared turbines
- 100 n.mi. design radius

in Figure VII-13 might have been expected, such that the retractable landing gear would pay off at the *higher* payloads and radii of action. However, when it is remembered that all geared turbine powered helicopters of this study which had payloads in excess of one ton were rotor-limited in cruise speed, it becomes apparent that little if any increase in block speed and work capacity (payload times block speed) resulted due to decreased drag *except* in the small size helicopter bracket (with low payload and radius of action). In the larger size bracket the only desirable effects were an insignificant reduction of fuel weight and percent power setting, and the main effects on 1/E were the increase in airframe weight and the attendant *weight growth* of all components. This weight growth was the predominant factor in forcing the somewhat unexpected trends in Figure VII-13.

Figure VII-14 is presented as a typical example of the comparative trends of 1/E versus design payload, for fixed landing gear and retractable landing gear, at an assumed radius of action of 100 nautical miles. For this radius of action, the cross-over point between the two landing gear types occurs at a payload of 3 tons. The boundary line between the two types shown in Figure VII-13 was established by the locus of several such cross-over points.

E. Effect of Payload Carried Externally

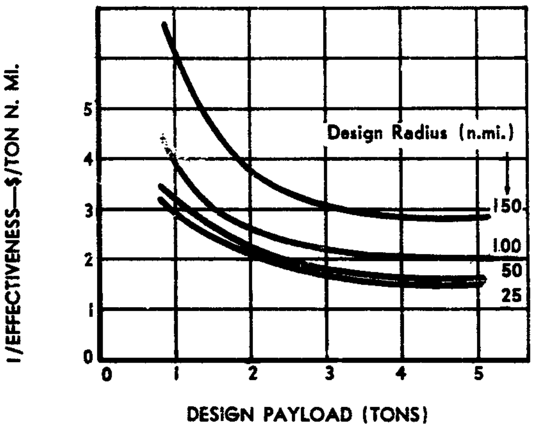
Figures VII-15 and VII-16 show the trends of 1/E versus design payload when the load is carried externally. These two examples, for geared gas turbine-powered single rotor and tandem rotor helicopters respectively, are for a design hover ceiling of 5000 feet. Separate curves are included for each of five design radii of action. By comparison of these curves with those for internal loads (Figures VII-6 and VII-8), it can be seen that externally carried payload produces lower values of 1/E than internally carried payload at 1) *low radii of action* where the reduction in block speed and increase in fuel required are of smaller significance, and 2) at *higher design payloads* where the elimination of loading and unloading time has greater significance. Figures VII-17 and VII-18 illustrate the payload-radius regimes in which one or the other of the two methods of carrying the payload produce the lowest value of 1/E. These trends and general regimes would be roughly the same for all helicopter types and design hover ceilings.

F. Effect of Increased Cruise Speed

The special analysis of the effect of increasing cruise speeds up to 130 knots at 5000 feet, based upon the supposition that compressibility drag divergence *may* create no significant vibratory problems, but only a rise in power required, is discussed in some detail in Chapter IV and Appendix B. The higher power requirements indicated that these high speed helicopters would exhibit standard atmosphere hover capabilities in excess of 7500 feet, and for this reason the special analysis was made for a design hover ceiling of 10000 feet, to insure adequate power for the required speeds.

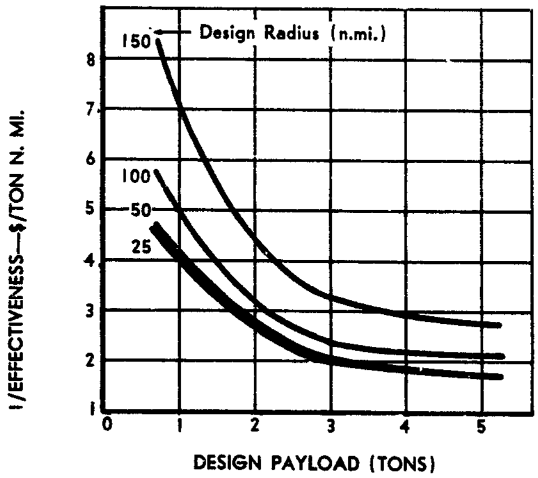
The results obtained are presented in Figure VII-19, which shows the variation of 1/E versus cruise speed for an example single rotor, geared gas turbine powered helicopter with a design payload of 4 tons. It is apparent from this chart that the cruise speed for minimum 1/E is in the order of 105 to 110 knots, accepting the inherent "state of the art" assumptions upon which the analysis was based. The penalty in cost per ton-nautical mile to be paid for higher cruise speeds up to 130 knots or

CHAPTER VII — EVALUATION — 1956 TO 1961



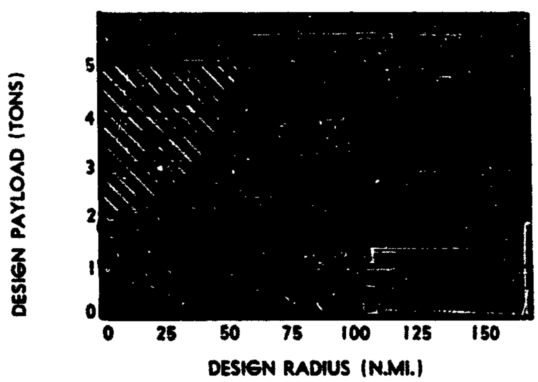
VII-15 EFFECT OF EXTERNALLY CARRIED PAYLOAD: COST/TON N.MI. VS. PAYLOAD FOR VARIOUS DESIGN RADII

- Single rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



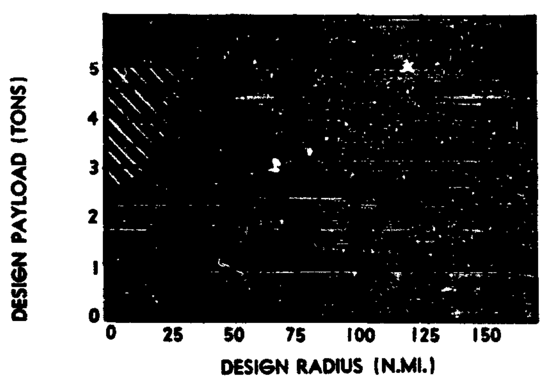
VII-16 EFFECT OF EXTERNALLY CARRIED PAYLOAD: COST/TON N.MI. VS. PAYLOAD FOR VARIOUS DESIGN RADII

- Tandem rotor helicopter
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



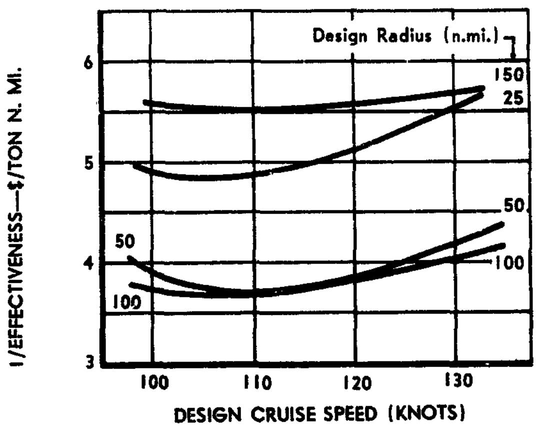
VII-17 DECISION CHART FOR EXTERNAL VS. INTERNAL LOAD

- Single rotor helicopter
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



VII-18 DECISION CHART FOR EXTERNAL VS. INTERNAL LOAD

- Tandem rotor helicopter
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



VII-19 EFFECT OF CRUISE SPEED: COST/TON N.MI. VS. DESIGN CRUISE SPEED

- Single rotor helicopters
- Twin geared turbines
- Design payload: 4 tons
- Utilization: 1200 hrs/year
- 10000' design hover ceiling, OGE, standard altitude

more is primarily a result of the predominance of maintenance cost as cruise percent power setting increases, offsetting the lesser increase in work capacity (ton-nautical miles per hour) attendant with the increased block speed.

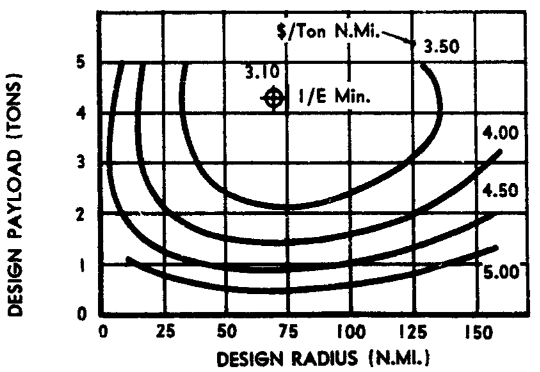
G. General Summary, 1956 to 1961

Figures VII-20 through VII-23 summarize the results of the basic 1956-1961 study which were covered in Section B. These charts were prepared by converting the computed results shown in Figures VII-5 through VII-8 into maps of design payload versus design radius of action, wherein the map contours represent constant values of cost per ton-nautical mile, 1/E. The best possible combination of design payload and design radius of action for each configuration analyzed is represented on these maps by a single point, and corresponding minimum value of 1/E, forming a unique saddle point around which the contours of higher 1/E values are located. This method of presentation facilitates the quick determination of 1/E for a selected design payload and design radius of action, and presents a general illustration of the state-of-the-art trends in cost per ton-nautical mile for transport helicopters envisioned for the 1956-1961 period. These four summary maps clearly illustrate two major conclusions for the 1956-1961 evaluation, namely:

1) With the exception of extremely high temperature and high altitude hover performance requirements together with high radius of action (in the order of 150 nautical miles as shown in Figure VII-12), the reciprocating engine powered helicopters are *not* competitive, since they exhibit 1/E values approximately twice as high as for the geared turbine powered helicopters. The geared gas turbine engine is therefore the optimum power plant selection for the 1956-1961 time period, with the exception of the one high temperature, high radius of action contingency as noted. The use of water injection, ground effect, and/or take-off power for the hover condition with geared turbine engines would effectively eliminate the reciprocating engine from further consideration for the *transport* mission.

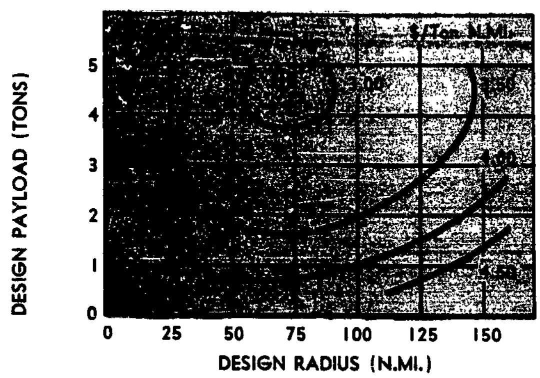
2) A choice between single rotor helicopters and tandem rotor helicopters cannot be made purely on the basis of the results presented herein, since the differences in 1/E between the two types are almost negligible, and are of the same order of magnitude as the accuracy of the study. However, the *payload* and *radius of action* which produce minimum 1/E for the tandem rotor helicopters were somewhat higher than for the single rotor helicopters, and this is a manifestation of the different *weight growth* trends exhibited by the two types.

CHAPTER VII — EVALUATION — 1956 TO 1961



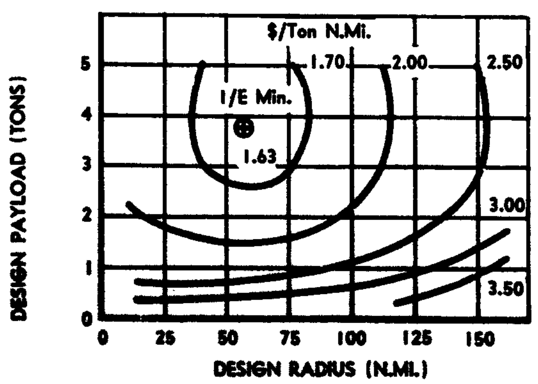
VII-20 DESIGN PAYLOAD VS. DESIGN RADIUS FOR VALUES OF COST/TON N.MI.

- Single rotor helicopter
- Twin reciprocating engine
- 5000' design hover ceiling, OGE, standard altitude



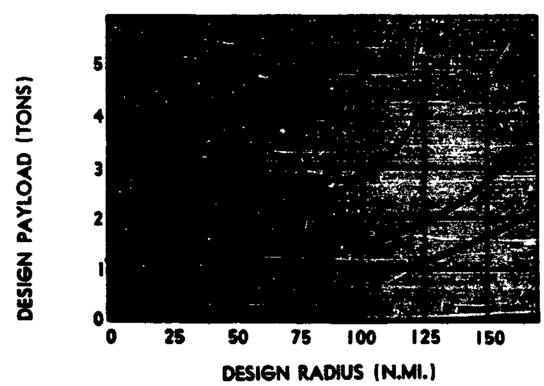
VII-22 DESIGN PAYLOAD VS. DESIGN RADIUS FOR VALUES OF COST/TON N.MI.

- Tandem rotor helicopter
- Twin reciprocating engines
- 5000' design hover ceiling, OGE, standard altitude



VII-21 DESIGN PAYLOAD VS. DESIGN RADIUS FOR VALUES OF COST/TON N.MI.

- Single rotor helicopter
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



VII-23 DESIGN PAYLOAD VS DESIGN RADIUS FOR VALUES OF COST/TON N.MI.

- Tandem rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude

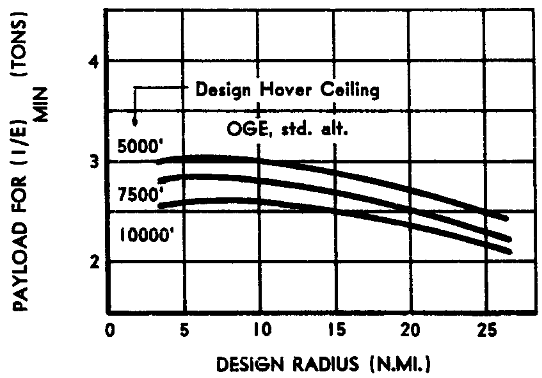
Chapter VIII
EVALUATION - 1960 to 1970

A. Transport Effectiveness Trends, Tip Power Plants

It is the purpose of this chapter to present the basic results derived from the parametric evaluation of helicopter configurations using *advanced tip power plants* currently in the developmental stage, and considered to be production possibilities in the time period 1960 to 1970. Specifically, *ramjet, tip-mounted turbojet,* and *pressure jet* power plant types were investigated as possibilities for this advanced time period, with their application limited to *single rotor* helicopters. The procedure used in processing the data for these advanced configurations was identical to that which has been outlined for the more conventional types with geared engines.

The general trends of 1/E with design payload, design radius of action, and design hover ceiling were found to be *qualitatively similar* to the trends discussed in Chapter VII, as expected. The familiar *weight growth* phenomena were again in evidence, as manifested by 1) the trend towards lower payload for minimum 1/E as radius of action was increased, 2) the increase in 1/E as hover ceiling was increased beyond a certain point, and 3) the minimization of 1/E at a unique combination of design payload and design radius of action.

Figures VIII-1 through VIII-3 show the variation of optimum payload (producing minimum 1/E) with design radius of action, for the ramjet, pressure jet, and tip-turbojet power plants. Curves are included on these charts for each of the three design hover ceilings considered.

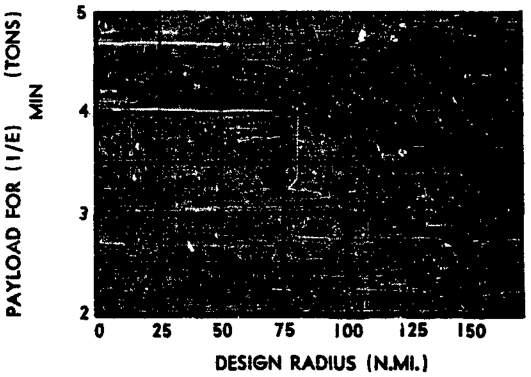


VIII-1 PAYLOAD FOR MINIMUM COST/TON N.MI. VS. DESIGN RADIUS FOR VARIOUS DESIGN HOVER CEILINGS
- Single rotor helicopter
- Tip mounted ramjet engines

Figures VIII-4 through VIII-6 show the typical trends of 1/E versus design payload, for various design radii of action, and for a design hover ceiling of 5000 feet as an example. The trends from these and similar charts for design hover ceilings of 7500 and 10000 feet were used to obtain the optimum payloads of Figures VIII-1 through VIII-3.

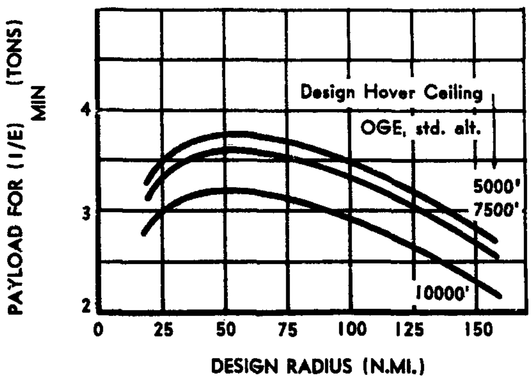
Note that the radii of action considered for the ramjet powered helicopters were very short, up to a maximum of 25 nautical miles. Analysis based on the ramjet "state of the art" assumptions used in this study, indicated that the cost of ramjet helicopters would be *pro-*

CHAPTER VIII — EVALUATION — 1960 TO 1970



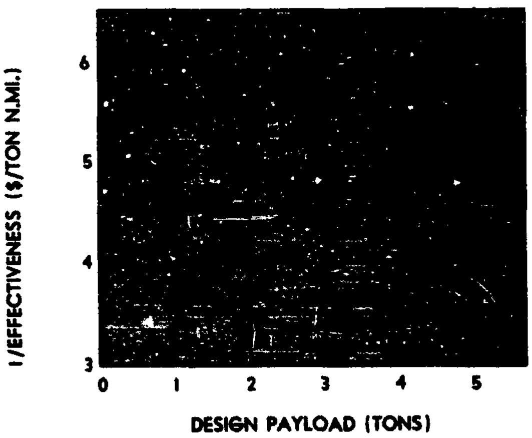
VIII-2 PAYLOAD FOR MINIMUM COST/TON N.MI. VS. DESIGN RADIUS FOR VARIOUS DESIGN HOVER CEILINGS

- Single rotor helicopter
- Pressure jet powerplant



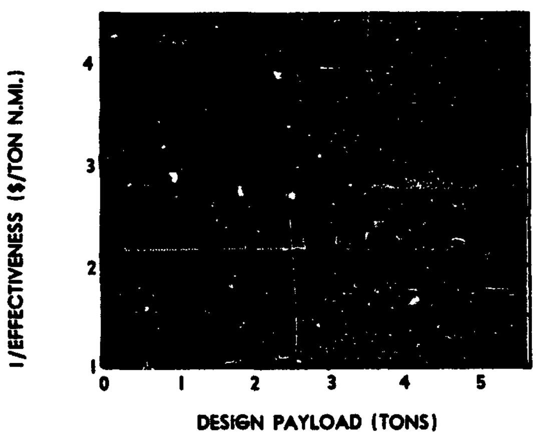
VIII-3 PAYLOAD FOR MINIMUM COST/TON N.MI. VS. DESIGN RADIUS FOR VARIOUS DESIGN HOVER CEILINGS

- Single rotor helicopter
- Tip mounted turbojet engines



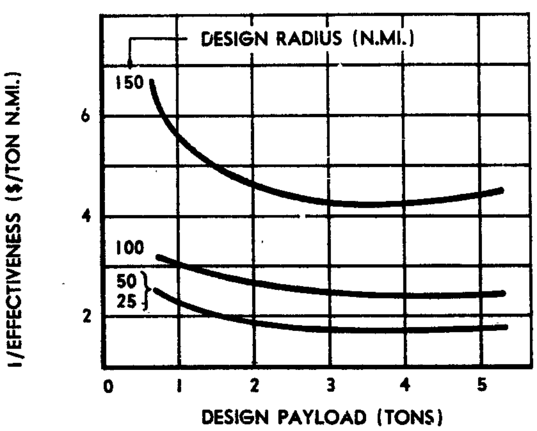
VIII-4 COST/TON N.MI. VS. DESIGN PAYLOAD FOR VARIOUS DESIGN RADII

- Single rotor helicopters
- Tip mounted ramjet engines
- 5000' design hover ceiling, OGE, standard altitude



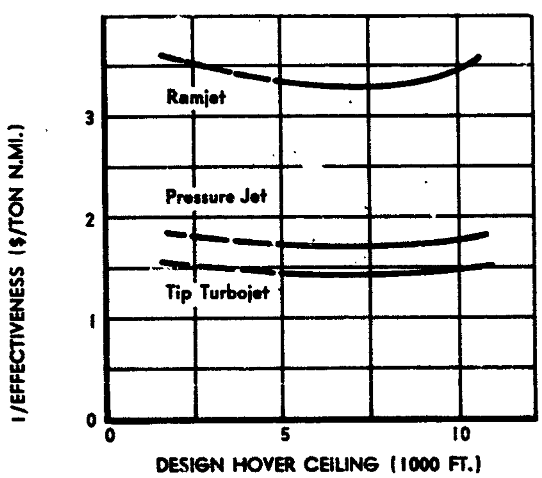
VIII-5 COST/TON N.MI. VS. DESIGN PAYLOAD FOR VARIOUS DESIGN RADII

- Single rotor helicopters
- Tip mounted turbojet engines
- 5000' design hover ceiling, OGE, standard altitude



VIII-6 COST/TON N.MI. VS. DESIGN PAYLOAD FOR VARIOUS DESIGN RADII

- Single rotor helicopters
- Pressure jet powerplant
- 5000' design hover ceiling OGE, standard altitude



VIII-7 COST/TON N.MI. VS. DESIGN HOVER CEILING, O.G.E.

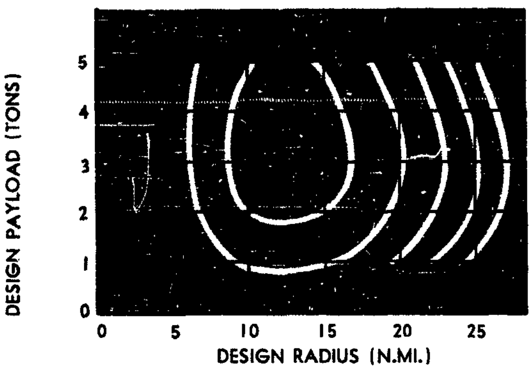
- Single rotor helicopters
- Tip mounted powerplants
- Cost per ton n.mi. based on optimum design payload and range for the hover ceiling

hibitively high at 50 nautical miles radius of action, and in fact the *weight growth* trends for this type appeared to preclude even the *possibility* of design radii of action greater than about 75 nautical miles with *no payload.* As a result, the ramjet powered helicopters could not compete on the basis of cost/ton n. mi., with any of the other types, including the geared-engines, at any except the *very short* radii of action. The tip turbojet and pressure jet helicopters, however, are shown to be strong competitors. Both of these types indicated values of 1/E in the same *general* order of magnitude as the geared gas turbine types discussed in Chapter VII.

B. Effect of Hover Ceiling

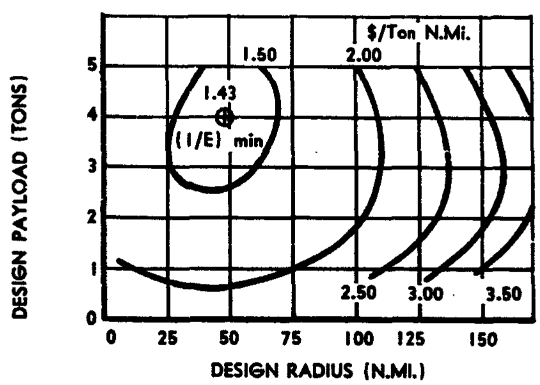
In Figure VIII-7, curves of 1/E versus design hover ceiling (in standard atmosphere, out of ground effect, using normal rated power) are shown for the three tip powered types. Similar to Figures VII-9 and VII-10 for the geared-engine types, these curves are made up of *hybrid* points of payloads and radii of action for *minimum* 1/E. The trends for all three types are nearly flat, indicating a moderate increase in 1/E above 7500 feet hover ceiling. This trend is quite similar to that noted for the geared gas turbine powered helicopters in Chapter VII, Figures VII-9 and VII-10, and is due to the fact that the *weight growth* with increased installed power is less predominant for these engine types than for reciprocating engines. A hover requirement of 6000 feet at 95°F would be the equivalent of a standard day hover ceiling in the order of 15000 to 18000 feet for these jet types, in which case more significant penalties in higher 1/E would be paid, similar to the trends shown in Figure VII-12 for geared turbine types.

CHAPTER VIII — EVALUATION — 1960 TO 1970



VIII-8 DESIGN PAYLOAD VS. DESIGN RADIUS FOR VALUES OF COST/TON N.MI.

- Single rotor helicopter
- Tip mounted ramjet engines
- 5000' design hover ceiling, OGE, standard altitude



VIII-9 DESIGN PAYLOAD VS. DESIGN RADIUS FOR VALUES OF COST/TON N.MI.

- Single rotor helicopter
- Tip mounted turbojet engines
- 5000' design hover ceiling, OGE, standard altitude

C. General Summary and Recommendations, 1960 to 1970

Figures VIII-8 through VIII-10 summarize the results of the 1960 to 1970 study in the form of maps of design payload versus design radius of action, with contours representing constant values of cost per ton-nautical mile, located around the unique minimum point corresponding to the best possible selection for each type. Inspection of these charts reveals clearly that 1) the best ramjet powered helicopter for the transport mission would have a radius of action of 10 to 15 nautical miles, and would produce a minimum 1/E of 3.27 dollars per ton-nautical mile at this radius, carrying a payload of 3 tons; and 2) this value of 1/E is not competitive with the minimum values shown on the charts for tip turbojet and pressure jet powered helicopters. Additional design summary charts for design hover ceilings of 7500 and 10000 are included in Appendix I.

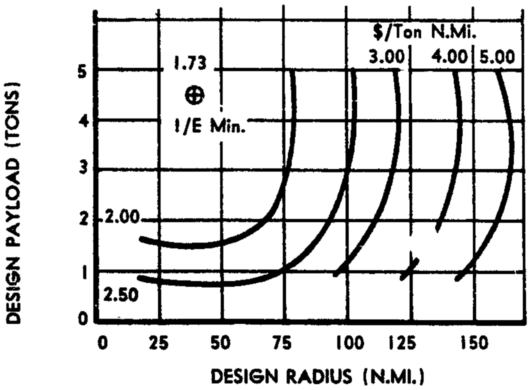
Before stating the conclusions and recommendations arising from these results of the 1960 to 1970 study, certain influential assumptions which are implicit in these results should be recapitulated:

1) Tip ramjet engines were assigned standard power, weight, and fuel consumption characteristics which might be termed slightly optimistic in the light of *current operational* engine characteristics, but which are justifiable estimates of expected improvements[1].

2) The tip turbojet engines were assigned *conservative* estimated characteristics from a Packard 1954 brochure[2] which has recently been superceded with more optimistic estimates, particularly with regard to fuel consumption rates. Should these later estimates prove to be attainable by the time period in question, the primary effect would be a further reduction in 1/E for the tip turbojet powered

[1] *Proposal for the Improvement of the Ramjet Engine for Helicopter Propulsion*; H.H. Report 545.3; November 30, 1954.

[2] *Helicopter Tip Turbojet Brochure*, Packard Motor Car Co., Aircr. Engr. Div. Report 7JE-103, September 27, 1954.



VIII-10 DESIGN PAYLOAD VS. DESIGN RADIUS FOR VALUES OF COST/TON N.MI.

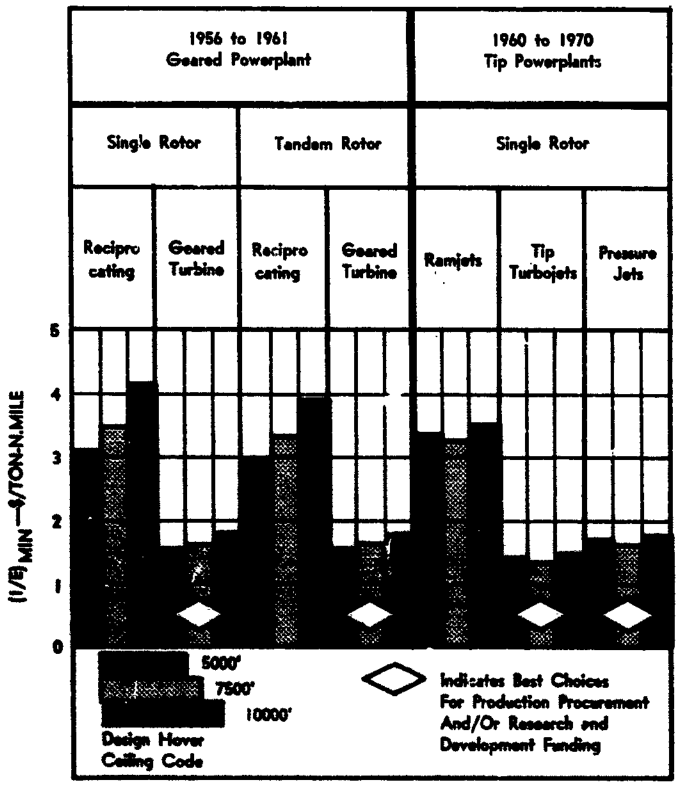
- Single rotor helicopter
- Pressure jet powerplant
- 5000' design hover ceiling, OGE, standard altitude

helicopters, and small increases in the optimum payload and radius of action.

3) The pressure jet power plants were restricted to the "cold cycle" type[1], since the "hot cycle" type which is of current interest is a complex system requiring in itself a very broad and thorough parametric analysis. For this present study, the compressed air was assumed to be supplied by a separate compressor, driven by a geared turbine engine with characteristics similar to those for direct geared power applications.

4) The effects of retractable landing gear,

[1] *Pressure Jet Power Plant Characteristics*, appended to *Design Analysis Methods* H. H. Report No. 473.6; November 30, 1955.



VIII-11 COMPARISON OF GEARED-ENGINE HELICOPTERS (1956 TO 1961) AND TIP-POWERED HELICOPTERS (1960 TO 1970) externally carried payload, and increased cruise speed were not analyzed for the advanced tip power types, since it is reasonable to expect that the general trends shown by these special considerations for the geared engine types, discussed in Chapter VII, would apply equally as well here.

Figure VIII-11 is a general summary chart which compares the results of the 1956-1961 basic study with those of the 1960-1970 study. The 1/E values are, as in Figures VII-9, VII-10, and VIII-7, made up of *hybrid* points for the optimum payload and radius of action, at each hover ceiling shown. Figure VIII-11, then, provides a pictorial justification for the following conclusions and recommendations:

1) Ramjet power does not appear to offer any competitive advantages, and extensive development programs for its application to helicopters for the ***transportation mission,*** in which the effectiveness criterion is *cost per ton-nautical mile,* cannot be recommended.

2) Both the pressure jet and tip turbojet power plant types appear to be quite competitive with the geared gas turbine type of power plant, and the tip turbojets indicate a slightly lower quantitative value of minimum 1/E than any other type of power plant considered. Since pressure jet powered helicopters are at the present time a reality, recommendations for their further development have a better foundation *in fact.* Lacking such solid foundation for the proposed tip turbojet power plants, a degree of caution must be implicit in the recommendations; however, further research and developmental design studies are an obvious recommendation for this type, especially in the light of the latest improvements in estimated fuel consumption.

Chapter IX
OPTIMUM HELICOPTER TRANSPORT SYSTEMS

A. Selection of Optimum Transport Systems

1. *Decision Analysis*

From the evaluation of the many interrelated variables and their effects on the military cost per ton-nautical mile, shown in Chapter VII, a final determination of optimum transport systems was possible. This final determination is presented in the *Decision Analysis* of Figure IX-1.

As shown in Chapter VII, all optimum systems considered within the hover ceiling matrix indicated gas turbine powerplants. Furthermore, since the difference in effectiveness between optimum single and tandem rotor types was negligible within the accuracy of the study, both types are encompassed by the curve of Figure IX-1 and are included in the characteristics table for optimum systems. Optimum systems for 5000 ft., 7500 ft. and 10000 ft. OGE standard day hover ceilings are each included in the decision analysis. For the variation in hover ceiling, the effectiveness differences for optimum systems were again negligible and are therefore included within the scatter band shown in the effectiveness chart of Figure IX-1. Optimum payloads, gross weights and required installed power vary not only with the configuration type, but also with hover ceiling. These variations are tabulated in the figure. If hover ceilings higher than those encompassed in Figure IX-1 are *positively* required, the optimum transport system must pay a rapidly increasing penalty in cost per ton-nautical mile, as discussed in Chapter VII.

In selecting an optimum system two factors must be ascertained. Namely: 1) Average mission radius of action expected between Army supply depots and combat element supply dumps, and 2) The average hover ceiling requirement consistent with expected future helicopter transport operations.

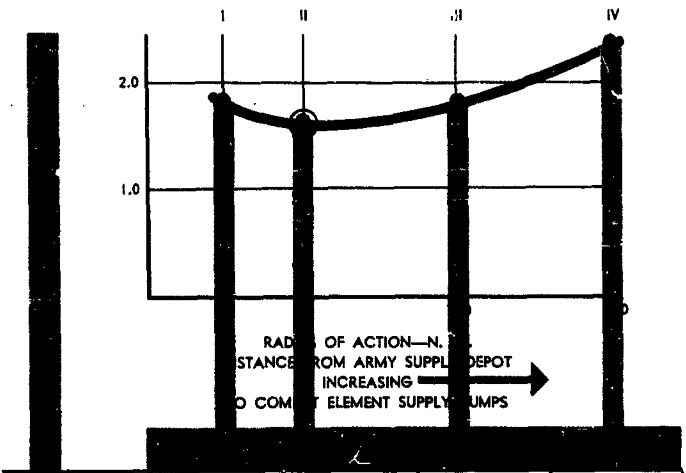
The hover ceiling requirement selection must naturally be based on mean operational conditions, but should also be viewed in terms of its influence on transport effectiveness.

With these two factors known the optimum Military Helicopter Transport System can be readily ascertained from Figure IX-1 whether the design configuration be single rotor or tandem rotor type.

As may be further seen from the decision analysis figure, the overall optimum system occurs at approximately 50 nautical miles radius of action. This decision ignores specific mission requirements, but emphasizes the fact that greater or lesser design radii of action entail inherent penalties in effectiveness. This minimum point in the effectiveness curve is characteristic, and essentially defines optimum helicopter "state of the art" and inherent technological and cost balances.

The design characteristics data of Figure IX-1 indicate higher optimum payloads for tandem types than for single rotor types for the same mission requirement, and an overall maximum payload variation of 3.2 to 4.5 tons

CHAPTER IX — OPTIMUM HELICOPTER TRANSPORT SYSTEMS



Configuration	Single	Tandem	Single	Tand.	Single	Tandem	Single	Tandem	Hover Ceiling OGE
Payload (Tons)	3.2	4.3	3.8	4.5	4.2	4.5	3.8	4.3	5000 Ft.
Gross Weight (Lbs.)	18500	22700	23700	26500	32000	32700	37700	39500	
Installed Horsepower	2300	2950	2840	3240	3350	3610	3810	4120	
Payload (Tons)	3.3	4.2	3.6	4.4	3.7	4.4	3.6	4.0	7500 Ft.
Gross Weight (Lbs.)	19500	23200	23300	27200	30000	33800	38000	40000	
Installed Horsepower	2850	3140	3040	3430	3570	3870	4350	4380	
Payload (Tons)	3.3	4.0	3.6	4.3	3.7	4.2	3.6	3.8	10000 Ft.
Gross Weight (Lbs.)	20000	23200	24000	27200	32600	34200	42000	42000	
Installed Horsepower	3170	3300	3310	3660	4020	4320	5030	5080	

NOTE: All systems are geared gas turbine powered

IX-1 DECISION ANALYSIS

for all possibilities. This is in keeping with the evaluation presented in Chapter VII.

2. *Penalty For Incorrect Decision*

The military planner, responsible for the procurement of efficient air transport systems with minimum military budgets, and faced with the determination of the average mission radius of action between Army supply depots and combat element supply dumps in possible future military conflicts, must rely on military intelligence information, global trends in diplomatic relations, and a considerable amount of intuition in estimating the probable dispersion of combat elements. If his estimate of the future military situation should prove to be incorrect, the concept of effectiveness penalty resulting from this error is of interest. With this penalty evaluation for an incorrect procurement decision at hand, the planner can then base his decision on the minimization of penalties for "off-optimum" operations.

The table of Figure IX-2 is presented to allow the estimation of *effectiveness penalty* in the event of an incorrect decision. This table results from the study of effectiveness variations of transport systems operating at "off-

[redacted]

	I	II	III	IV
I	0	Negl.	5%	15%
II	5%	0	Negl.	10%
III	10%	Negl.	0	5%
IV	15%	5%	Negl.	0

IX-2 APPROXIMATE PENALTY FOR INCORRECT DECISION

design" payloads and radii of action.

Inspection of Figure IX-2 shows that the penalties do not exceed 15%, even in the most extreme cases. In general, this is due to the rather flat character of the cost per ton-nautical mile vs. payload curves over the scope of optimum payloads. As may be seen, Systems I or IV, representing 25 and 150 n.mi. design radii of action respectively, suffer increasing degrees of effectiveness penalty as the actual operation diverges from the System design point. When System II is operated under the conditions which would make System I an optimum choice, a 5% penalty results, and when System IV is operated under conditions which would make System III an optimum choice, again a 5% penalty is found.

Figure IX-2 indicates, then, that Systems I or IV will suffer a significant penalty at any of the "off-design" point conditions shown in the decision analysis of Figure IX-1. On the other hand, Systems II and III suffer negligible penalty under "off-design" operation, except under extreme conditions, in which case the penalties are only 5%.

If the military planner can make the assumption that the situations indicated by Systems I through IV all have equal probability of occurrence, then the procurement decision can be based on the system which minimizes the penalty for all conditions. This would show either System II or III as best. Reference to Figure IX-1, however, indicates that System II would be the optimum choice since it produces a lower total cost per ton-n.mi. than System III while indicating the same degree of effectiveness penalty under "off-design" point conditions.

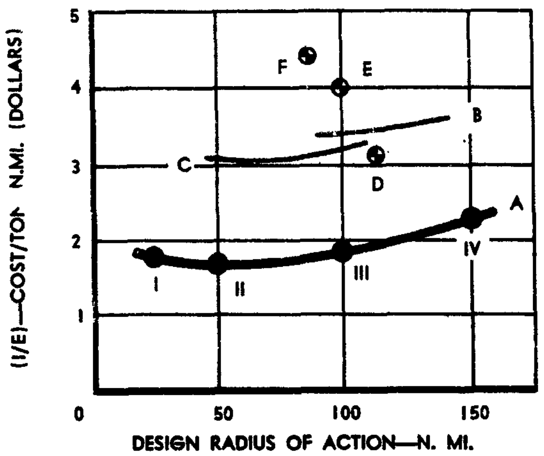
However, if the situation indicated by any one of the optimum systems is felt to have a higher relative probability, then the best choice must reflect a weighted consideration of both penalty and probability of occurrence, and the optimum choice could be determined by minimizing the mathematical product of effectiveness penalty and occurrence probability. For example, in comparing Systems II and III, if operation under the conditions of System IV were considered to be more probable than operation under the conditions of System I, then the best choice would shift from System II to System III. An incorrect decision, however, between Systems II and III would never result in more than a 5% effectiveness penalty.

B. Comparison With Presently Possible Systems

In viewing the optimum indications from this study in the light of helicopter transport systems which are now possible, or at least proposed, the following comparisons are presented.

Figure IX-3 shows the various trends and calculated points of military cost per ton-n.mi.

CHAPTER IX — OPTIMUM HELICOPTER TRANSPORT SYSTEMS



A—Optimum transport system indications for 5000, 7500 and 10000 ft. OGE hover ceiling; all indications — turbine powered

B—Predicted trend for single rotor, reciprocating powered helicopters—5000 ft. OGE hover ceiling. Avg. % NRP in Cruise = 50%

C—Predicted trend for tandem rotor, reciprocating powered helicopters—5000 ft. OGE hover ceiling. Avg. % NRP in Cruise = 70%

D—Calculated point from military specifications, estimated weight and performance data on H-37 single rotor, reciprocating powered helicopter—% NRP in Cruise = 45%

E—Calculated point for optimum helicopter meeting proposed AGF specification hover ceiling 6000 ft. OGE @ 57° over standard temperature

F—Calculated point from military specification, estimated weight and performance data on H-16A tandem rotor, reciprocating powered helicopter—% NRP in Cruise = 80%

IX-3 COMPARISON OF PRESENT AND PROPOSED TRANSPORT HELICOPTERS WITH OPTIMUM INDICATIONS—1956 to 1961

versus design radius of action. Curve (A) represents the effectiveness trend of optimum systems resulting from this study. These systems are all comprised of turbine powered configurations and exhibit hovering performance from 5000 ft. to 10000 ft. OGE standard day.

Curve (B) represents the predicted trend for single rotor reciprocating powered types as shown in Chapter VII. The helicopters indicated by curve (B) cruise at 50% of normal rated power on an average, while the tandem, reciprocating powered helicopters, indicated by curve (C) cruise at an average of 70% normal rated power.

Point (D) represents a cost per ton-n.mi. calculated from military data for the H-37 helicopter, assuming that its operating capabilities meet its design specification. As may be seen, this point falls slightly below the predicted trend (B). This is primarily due to 1) its lower percentage (45%) normal rated power in cruise, which gives a lower maintenance cost on transmissions and drives and engines, and 2) its slightly lower weight compared to the predicted weight corresponding to curve (B) which lowers both maintenance and depreciation cost. It does not approach, however, the effectiveness of the optimum turbine powered helicopters indicated by curve (A).

Point (F) represents a cost per ton-n.mi. calculated from military data for the H-16A tandem rotor, reciprocating powered helicopter, assuming that its operating capabilities meet its design specification. The design hover ceiling at normal rated power for this helicopter was calculated, using military design specification data, to be under 1000 ft. OGE, standard day. It may be seen that this point falls considerably above the predicted trend (C) for tandem rotor reciprocating powered helicopters. This is primarily due to its high percentage of normal rated power required in cruise, which has a serious effect in increasing the transmission and drives and engine maintenance costs.

Point (E) represents a cost per ton-n.mi. calculated for an Army ground forces proposed specification[1] for a 3 ton payload heli-

[1] Letter No. 24356 from Office, Chief of Army Field Forces, to Assistant Chief of Staff, G-3 "Military Characteristics for a 3-Ton Helicopter"—21 January 1954.

copter of the indicated design radius of action and having hover performance of 6000 ft. OGE at 95°F. This helicopter was reciprocating powered, since for this extremely high hover performance, a reciprocating powerplant gave a lower gross weight than did a turbine powerplant. As may be seen, this helicopter is less efficient than the helicopters represented by the predicted trend (C) for reciprocating powered tandem rotor types having standard day hover ceilings from 5000 ft. OGE, to 10000 ft. OGE. Furthermore, it indicates only a slightly lower cost than that indicated for the H-16A helicopter which has very low hover performance. This is due to the extremely high gross weight penalty which is brought about by the high hover performance requested in the proposed specification. An inspection of the cost per ton-n.mi. versus hover ceiling curves, shown in Chapter VII, will explain why helicopters having *very low or very high* hover performance display high cost of operation. If such a high degree of hover performance is mandatory, then the point (E) represents an optimum possibility. However, it would constitute a 110% penalty in transport effectiveness, or more than twice the military budget indicated by Optimum System III of this study.

It may therefore be seen from Figure IX-3 that future optimum Military Helicopter Transport Systems can induce total costs as low as one half to two thirds the cost of presently operational or suggested possibilities provided 5000 ft. to 10000 ft. OGE standard day hover ceilings will permit satisfactory operation under the majority of temperature-altitude conditions.

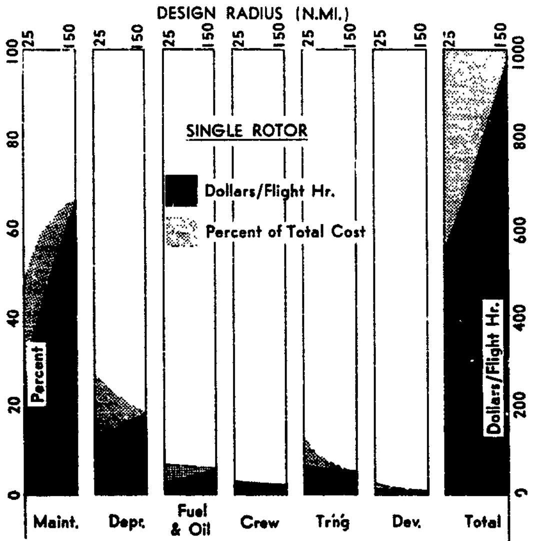
C. Cost Analysis For Optimum Systems

The transport system cost, calculation methods for which have been discussed in Chapter VI, were calculated as required by the effectiveness measure for all system possibilities. The cost breakdowns expressed in dollars and percentages of total cost are shown in Figures IX-4 and IX-5 for optimum systems I, II, III and IV, for a design hover ceiling of 5000 ft. OGE, standard day. The data, however, applies as well to 7500 ft. and 10000 ft. OGE, standard day hover ceilings, since the cost differences between the three hover ceilings are negligible.

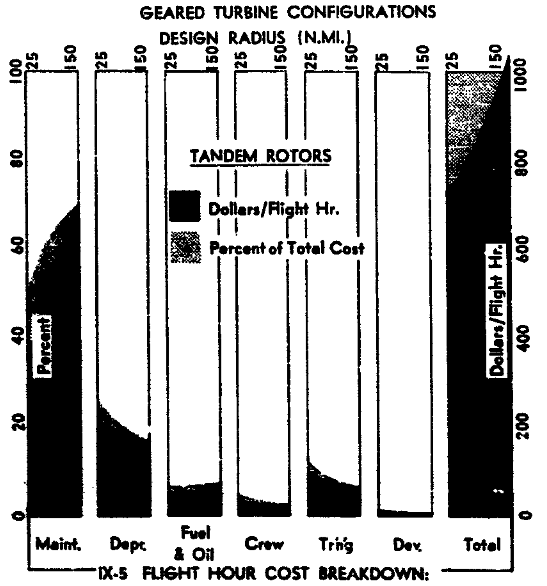
The single rotor types for optimum systems displayed an average power setting in cruise of 60% NRP, while the tandem types indicated an average of 74%. The tandem configuration, being more efficient in hovering, requires less installed power per pound of gross weight for a given hover requirement than a single rotor type. It follows that the tandem type will cruise at higher percentages of its installed power for equal drag considerations. Since the maintenance cost on transmissions and drives and engines increases with the percent normal rated power required in cruise, the maintenance cost for tandem types was, in all cases, greater than for the single rotor types. However, since the tandem types indicated optimum systems at higher payloads than did the single rotor types, the increased tandem cost was offset by the increased payload or work capacity, and the costs per ton-n.mi. for the two types were essentially equal.

Another factor affecting the costs was aircraft utilization. Since, on the basis of a fixed aircraft availability, the utilization was found as a function of loading and unloading time and therefore of payload, the aircraft utilizations for the optimum systems ranged from 700 to 1300 hours per year. Greater utilizations would, of course, lower the total operating cost.

CHAPTER IX — OPTIMUM HELICOPTER TRANSPORT SYSTEMS



IX-4 FLIGHT HOUR COST BREAKDOWN: GEARED TURBINE CONFIGURATIONS



IX-5 FLIGHT HOUR COST BREAKDOWN: GEARED TURBINE CONFIGURATIONS

As may be seen from Figures IX-4 and IX-5, maintenance costs for all systems and types averaged 59% of the total cost. Depreciation costs were next in magnitude at an average of 23% of the total cost. The remaining 18% was made up as follows: Training, 8%; Fuel and Oil, 6%; Crew, 2½%; and Development, 1½%. Total flight-hour costs ranged from $550 for the smallest helicopter to $1030 for the largest.

D. Effect of Powerplant Availability . . . 1956 to 1961

As discussed previously in Chapter IV, all configurations investigated in the study were assigned generalized powerplant characteristics and no consideration was given to whether or not the required available power could be obtained in the form of production engines during the assigned time period. Having arrived at possible optimum transport systems, it becomes advisable to investigate the powerplant possibilities for procurement during the period 1956 to 1961. These are illustrated in Figure IX-6, which shows the proximity to optimum systems of configurations which could be built with existing engines.

1. *Available Powerplants*

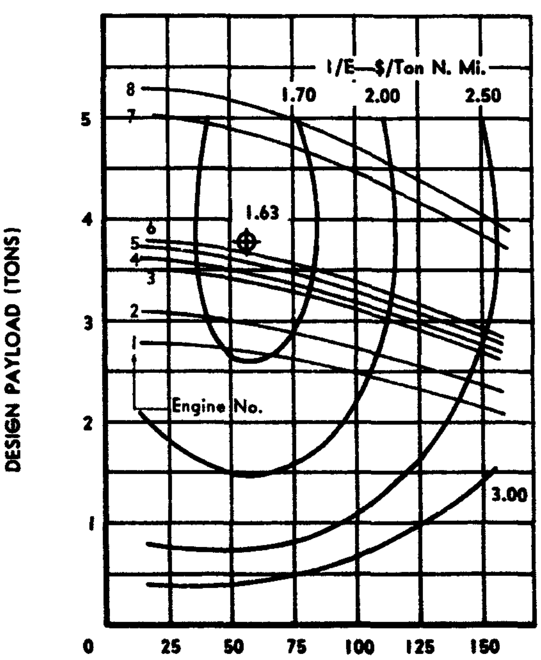
Since all optimum indications were for geared turbine powerplants, turbine engines of applicable size which are currently anticipated to be available by mid-1958 are shown in Figure IX-6. As may be seen, Figure IX-6 includes British, French and American engines. However, all of the foreign designed powerplants shown either have or are expected to have U.S. industry manufacturing licences by mid-1958.

2. *Powerplant Availability Effect on Optimum System Possibilities*

In Figures IX-7 through IX-12 contours of constant cost per ton-n.mi. are presented on a payload-radius chart. Superimposed on these charts are curves representing possible helicopter configurations for each powerplant model listed in Figure IX-6. The charts are shown for both single rotor and tandem rotor

ENGINE MANUFACTURER	MODEL	NORMAL RATED POWER	POSSIBLE AMERICAN MANUFACTURER	CODE NUMBER
Wright	TP43A1	1010		1
Rolls Royce	505 Dart	1120	Westinghouse	2
Rolls Royce	605 Dart	1290	Westinghouse	3
Lycoming	T-55	1325		4
Rolls Royce	R Da5	1375	Westinghouse	5
Armstrong Siddeley	GT-43 Mamba	1410	Wright	6
Turbomecca	GABIZOS	1940	Continental	7
Napier	N. EL-1 Eland	2072		8
Allison	T-56	2880		9

IX-6 POSSIBLE GEARED TURBINE POWERPLANTS AVAILABLE—1956 to 1961



IX-7 CONTOURS OF CONSTANT EFFECTIVENESS AND HELICOPTER DESIGN TRENDS FOR GIVEN POWERPLANTS

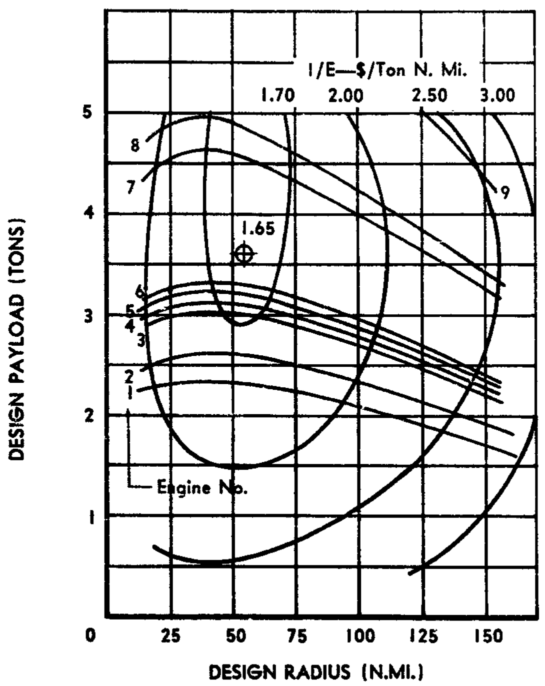
- Single rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude

types and for design hover ceilings of 5000 ft., 7500 ft. and 10000 ft. OGE, standard day. Upon inspection of these curves it may be seen that minimum cost possibilities can be selected from the available powerplant curves on the basis of minimum distance from the contour for lower cost per ton-n.mile.

If the best points for available powerplants are then compared with the characteristics data for optimum systems shown in Figure IX-1, the following pertinent facts become apparent. 1) The values for cost per ton-n.mi. for the possible aircraft, considering powerplant availability, fall within the scatter band shown in Figure IX-1 with only few exceptions; and these few exceptions occur at high hover ceilings combined with high radius of action. An optimum payload shift, either up or down is indicated, however, in almost every case. The shifts in payload are from 12% to 15%, with one or two as high as 20% and 25%. The effectiveness remains essentially the same as for the optimum systems discussed previously due to the fact that the cost per ton-n.mi. vs. payload curves for all configurations have a flat characteristic in the range of higher payloads. It may be concluded, therefore, that the powerplants considered to be available by mid-1958 will not force severe penalties in transport effectiveness from the ideal optimum systems. It may be further noted that the power plant availability consideration will shift the design payload either up or down from 12% to 25% of its optimum value without causing appreciable transport effectiveness penalties.

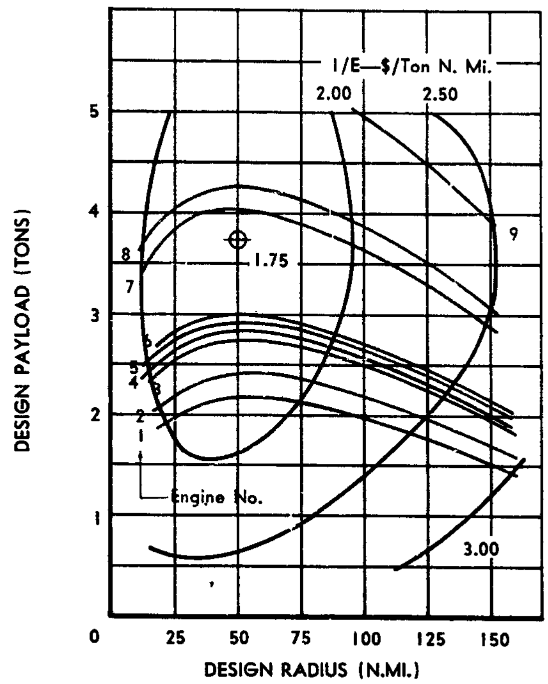
If the powerplants are selected on the basis of minimizing the shift in payload from that indicated for an optimum system, for the two rotor system types and the three hover ceilings included in the data, the following selections

CHAPTER IX — OPTIMUM HELICOPTER TRANSPORT SYSTEMS



IX-8 CONTOURS OF CONSTANT EFFECTIVENESS AND HELICOPTER DESIGN TRENDS FOR GIVEN POWERPLANTS

- Single rotor helicopters
- Twin geared turbines
- 7500' design hover ceiling, OGE, standard altitude



IX-9 CONTOURS OF CONSTANT EFFECTIVENESS AND HELICOPTER DESIGN TRENDS FOR GIVEN POWERPLANTS

- Single rotor helicopter
- Twin geared turbines
- 10000' design hover ceiling, OGE, standard altitude

are indicated for the optimum systems.

Engine No. 6—Mamba—8 possible applications.

Engine No. 7—Turbomecca—10 possible applications.

Engine No. 8—Napier Eland—4 possible applications.

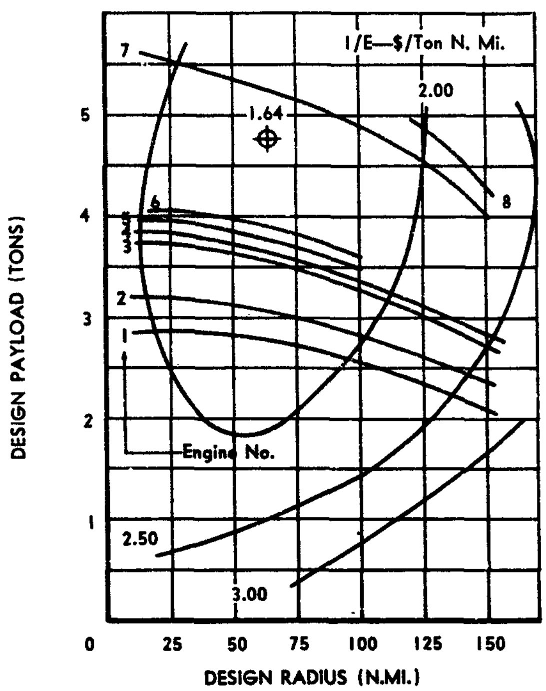
Engine No. 9—Allison T-56—2 possible applications.

Engine No. 9 applications occur only for the 10000 ft. hover ceiling and at 150 n.mi. radius of action, for System IV.

Engine No. 8 applications occur only for System IV, except for a 10000 ft. hover ceiling when the application falls to System III for both tandem and single rotor types.

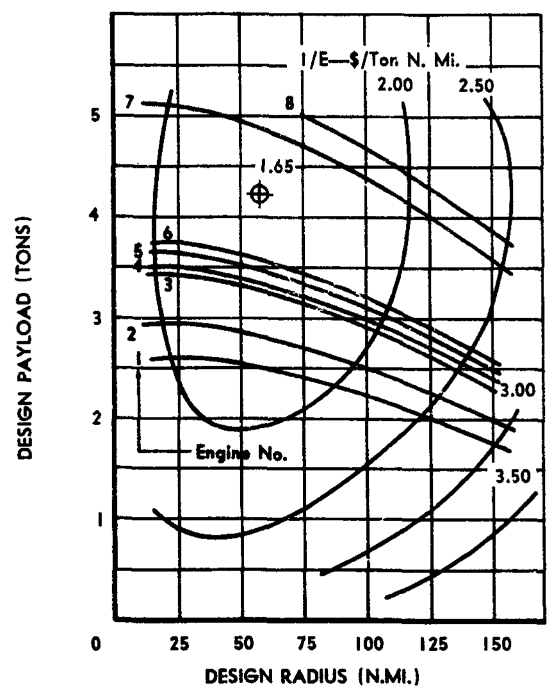
Applications of engines No. 6 and No. 7 occur throughout the hover ceiling range and for both rotor system types, for Systems I, II and III.

Since engines No. 6 and No. 7 have the highest potential applicability, it would appear that an engine midway between these two rated at 1650 hp, would be optimum. If procurement of Systems I through IV is contemplated, accelerated development of engines No. 6 through No. 9 is recommended so that early production can be realized.



IX-10 CONTOURS OF CONSTANT EFFECTIVENESS AND HELICOPTER DESIGN TRENDS FOR GIVEN POWERPLANTS

- Tandem rotor helicopters
- Twin geared turbines
- 5000' design hover ceiling, OGE, standard altitude



IX-11 CONTOURS OF CONSTANT EFFECTIVENESS AND HELICOPTER DESIGN TRENDS FOR GIVEN POWERPLANTS

- Tandem rotor helicopter
- Twin geared turbines
- 7500' design hover ceiling, OGE, standard altitude

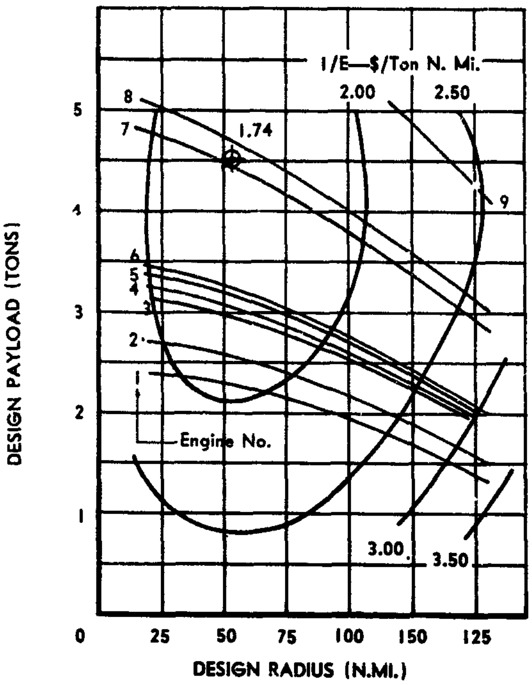
E. Military Force Requirement Estimates

As previously discussed in Chapter III, force requirements can be predicted only if the airlift tons required and the airlift distance involved can be accurately determined. After establishment of force requirements for a given combat element, the scale-up to more and larger elements must be conducted with great caution since the sc le-up is not linear unless the same military situation for all elements can be assumed and unless the airlift requirement for all elements is identical.

In an effort to appraise a typical set of force requirements, so that some insight as to the relative magnitudes involved could be obtained, the military force requirements for the isolated infantry division as depicted in Figure III-1 were determined and are presented in Figure IX-13.

These force requirements are presented for the optimum systems as described in Figure IX-1, and it may be seen that the daily cost of support of the division varies from about $55,000 to $470,000, depending on hover ceiling requirement and mission radius of action. These costs can be derived for any transport system regardless of airlift tonnage re-

CHAPTER IX — OPTIMUM HELICOPTER TRANSPORT SYSTEMS



IX-12 CONTOURS OF CONSTANT EFFECTIVENESS AND HELICOPTER DESIGN TRENDS FOR GIVEN POWERPLANTS

- Tandem rotor helicopter
- Twin geared turbines
- 10000' design hover ceiling, OGE, standard altitude

quirement, mission radius of action or particular aircraft involved, from the following relationship:

$$\text{Total Daily Cost} = TR \times (1/E)$$

where T = airlift tons per day required by a particular combat element

R = average distance between an Army supply depot and the particular combat element (nautical miles)

(1/E) = total military cost per ton-nautical mile for the particular aircraft system being investigated.

The value for 1/E must, of course, be consistent with the value for R which is used.

Figure IX-13 also shows that the number of aircraft required to support the single isolated division varies from about 50 to 130 depending, for the most part, only on mission radius of action. The number of aircraft required can be determined for any set of conditions from the following relationship.

$$N_S = \frac{T}{A}\left(K_I + \frac{R}{PV_B}\right)$$

This equation is derived in Appendix A and the new symbols shown are as follows:

N_S = Number of aircraft required

A = Average aircraft daily availability in hours

K_I = Cargo loading and unloading rate (hrs./ton) (assumed as .266)

P = Allowable payload per aircraft operating over radius R in tons

V_B = Mission block speed in knots

The number of maintenance personnel required, shown in Figure IX-13, was determined from Figure VI-8 and includes all personnel for not only first and second echelon maintenance, but also depot and overhaul phases.

Considering a 2% per month operational attrition rate, anywhere from 1 to 3 additional aircraft per month of operation are indicated as being required by Figure IX-13. The costing of these additional aircraft is not included since the costs of operational attrition are primarily related to the overall support and pipeline problems between the zone of the interior and a particular theatre of operations, *which have not been a part of this study*. The normal

HOVER CEILING OGE STANDARD DAY (FT)	HELICOPTER TYPE (ALL TURBINE POWERED)	DESIGN RADIUS OF ACTION (N MILES)	OPTIMUM PAYLOAD (TONS)	NUMBER OF SHIPS REQUIRED	NO. OF MAINTENANCE PERSONNEL REQUIRED	TOTAL COST PER DAY ($1000/DAY)	TOTAL FUEL PER DAY (GAL/DAY)	ATTRITION REPLACEMENT SHIPS/MO.
5000	SINGLE ROTOR,	25	3.2	53	558	54.8	27100	1
		50	3.8	64	723	100.5	45000	1
		100	4.2	76	949	225.7	91400	2
		150	3.8	123	1612	440.0	159200	2
	TANDEM ROTOR,	25	4.3	48	533	55.5	24200	1
		50	4.5	59	688	100.5	42600	1
		100	4.5	84	1040	225.7	92000	2
		150	4.3	113	1493	406.5	168600	2
7500	SINGLE ROTOR	25	3.3	52	558	55.5	27900	1
		50	3.6	67	756	100.5	46700	1
		100	3.7	95	1168	231.8	99600	2
		150	3.6	127	1680	443.0	182900	3
	TANDEM ROTOR,	25	4.2	48	542	56.7	26800	1
		50	4.4	60	705	100.5	45800	1
		100	4.4	85	1082	225.7	99000	2
		150	4.0	117	1567	431.0	183700	2
10000	SINGLE ROTOR,	25	3.3	52	573	56.1	30250	1
		50	3.6	67	773	106.6	50000	1
		100	3.7	96	1220	253.8	109150	2
		150	3.6	127	1758	446.0	204200	3
	TANDEM ROTOR,	25	4.0	49	548	58.5	30350	1
		50	4.3	61	720	105.3	52000	1
		100	4.2	88	1120	238.0	113100	2
		150	3.8	124	1680	465.0	216300	2

NOTE: Based on supply requirements of an isolated infantry division in assault of a prepared position. Total tonnage required is 610 tons per day.

IX-13 FORCE REQUIREMENTS FOR OPTIMUM SYSTEMS

flight hour charges for these additional aircraft cannot be assumed, since only the required number for the mission will be actually operating at any one time.

Chapter X
CONCLUSIONS & GENERAL RECOMMENDATIONS

It is believed that the objectives set forth in Chapter I have been met in this study and that the interrelated effects of technical, economic and operational parameters have been examined and presented in sufficient detail to allow the quantitative appraisal and selection of optimum Military Helicopter Transport Systems. The following general conclusions or recommendations can be summarized.

1) Optimum system selection is not a function of airlift tonnage required and may therefore be based on the effectiveness of a single helicopter configuration possibility.

2) Optimum system selection is not a function of the details of possible airlift route structure and may therefore be based on the expected average radius of action between an Army supply depot and combat element supply dumps.

3) Optimum system selection is a function of the relative rates of change of transport work capacity (ton-n.mi./hr.) and total cost, and maintenance cost has a major effect on the selection.

4) Available powerplants do not appreciably alter the design characteristics from those indicated for optimum systems, and have a negligible effect on changing the system effectiveness or cost per ton-n.mi.

5) Geared gas turbine powered helicopters are indicated for all optimum systems for the 1956 to 1961 time period.

6) Tandem and single rotor types indicate almost identical values for transport effectiveness but tandem types optimize at higher design payloads.

7) Optimum design payloads fall between 3.2 and 4.5 tons and vary with helicopter configuration, powerplant type and design hover ceiling.

8) Helicopters with very high payloads do not provide optimum transport systems since they are penalized, due to loading time, in the amount of airlift they can provide in a fixed number of available hours.

9) Increases in design hover ceiling beyond 10000 ft., OGE, standard day, induce substantial increases in cost per ton-n.mi. which approach the ton mile costs of helicopters having very low hovering performance.

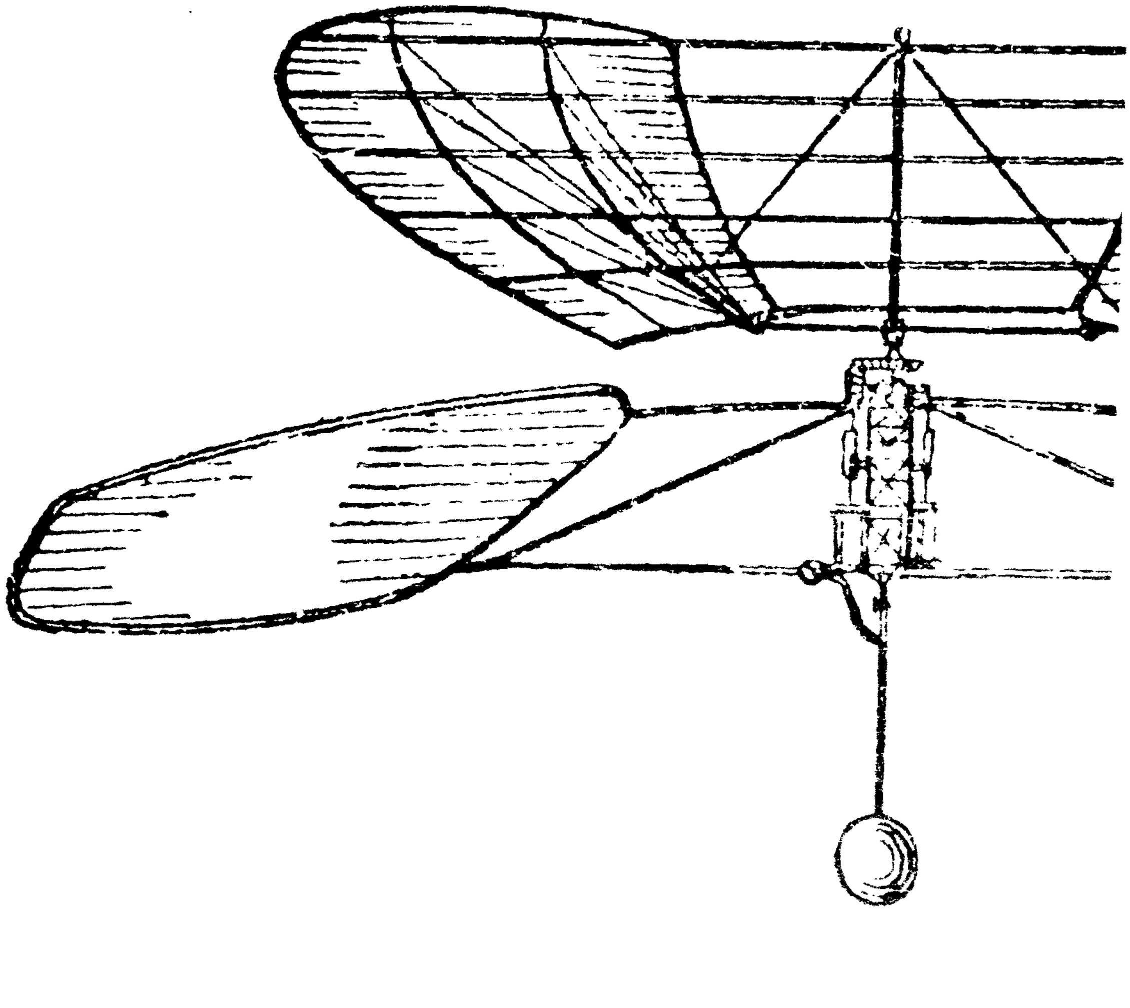
10) The ideal helicopter transport system, which occurs for a mission radius of action of 50 n. mi., indicates a total military cost per ton-n.mi. of $1.63, for the time period 1956 to 1961.

11) Tip powered transport helicopters of the pressure jet and tip turbojet types show promise for the future (1960 to 1970) in lowering ton-mile costs.

The quantitative information contained in the chapters and appendices of this report should provide sufficient background to allow the overall appraisal of transport helicopters applied to Army logistic transport missions, and in addition, allow the proper selection of the many variables involved to provide optimum helicopter transport systems for the time period, 1956 to 1961. In addition, indications for optimum advanced types are presented as a guide for research and development funding during the same time period, and possible production procurement by 1970.

The study has not attempted to evaluate helicopter transport systems as the most effective Army logistic support tool nor has it compared transport helicopters with other ground or air vehicles for the logistic transport mission. Both of these questions should be studied and answered by suitable investigation before production procurement of optimum helicopter transport systems for Army logistic support missions is implemented. The gain from such investigations may be very large and their cost is small. For example, this study was completed at a cost of approximately 14% of the cost of a single optimum transport helicopter.

SUPPORTING DATA



Appendix A

DERIVATION OF THE MEASURE OF EFFECTIVENESS

Derivation of the Measure of Effectiveness

Consider the military transport mission in terms of vehicle capability and total operational cost expenditure. A pertinent criterion from these considerations, constituting a true measure of transport efficiency or effectiveness, is:

$$E = \frac{\text{Total Military Cost per Hour}}{\text{Ton-Nautical Miles Transported per Hour}}$$

or

$$E = \frac{\text{Ton-Nautical Miles}}{\text{Military Dollar}} = \frac{T\,R}{C\,U_D\,N_S}$$

where

- T = Total tons per day required
- R = Radius of action or stage length (N.Miles)
- C = Total cost per flight hour (dollars)
- U_D = Daily aircraft utilization (flight hours)
- N_S = Number of aircraft required

N_S, the number of aircraft required to meet a given daily airlift requirement, T, in A available hours is derived as follows:

$$\text{Required average tons per hour} = \frac{T}{A}$$

where A = aircraft availability = flight time + loading time.

Available tons per aircraft hour =

$$\frac{P}{t_F + t_l}$$

where

- P = Payload per aircraft trip (tons)
- t_F = Flight time per aircraft trip (hours)
- t_l = Load and unload time per aircraft trip (hours)

But $t_F = \dfrac{R}{V_B}$

and $t_l = K_l P$

where

- V_B = Trip block speed (knots)
- K_l = Loading and unloading rate (Hrs./ton)

which gives

Available tons per aircraft hour =

$$\frac{1}{K_l + \dfrac{R}{PV_B}}$$

N_S is found as the quotient of the required tons per hour, and the available tons per aircraft hour, or

$$N_S = \frac{T}{A}\left(K_l + \frac{R}{PV_B}\right)$$

or

$$N_S = \frac{T}{A}\left(\frac{R}{PV_B}\right)\left(1 + K_l P V_B/R\right)$$

Now substituting the value for N_s into the effectiveness equation gives

$$E = \frac{PV_B}{C}\left(\frac{A}{U_D}\right)\left(\frac{1}{1 + K_I PV_B/R}\right)$$

Inspection of the above equation indicates that the term PV_B/C actually represents the ton-miles per hour, while the remaining portion is an identity which must be satisfied as:

$$\frac{A}{U_D}\left(\frac{1}{1 + K_I PV_B/R}\right) \equiv 1$$

It may be seen that the nume::ator and denominator of the above equatio.. represents the total daily availability; the denominator in terms of the daily utilization and the ratio of loading time to flight time and the numerator, directly.

The equation insists that for a fixed aircraft utilization, the availability must vary with payload or loading time, and that for a fixed aircraft availability the utilization must be found as a function of the ratio of the loading time to flight time.

Since a portion of the total flight hour cost (C) is a function of aircraft utilization, the proper utilization for each design configuration was determined by the following equation:

$$U_D = \frac{A}{1 + K_I PV_B/R}$$

Inverting the effectiveness criterion to the more common form for transport evaluation (cost/ton-n.mi.) and consideraing a route structure having Θ stage lengths, individually designated by j gives:

$$\frac{1}{E} = \frac{\sum_{j=1}^{\Theta} c_j \left(\frac{Tt}{P}\right)_j}{\sum_{j=1}^{\Theta} \left(TR\right)_j}$$

Where

t = total flight time required per aircraft per trip

$\left(\frac{Tt}{P}\right)_j$ = total flight hours per day required to airlift T tons over jth stage length

When a single radius or range mission is considered $\Theta = 1$

and

$$\frac{1}{E} = \frac{C\left(\frac{Tt}{P}\right)}{TR}$$

But since $(R/t) = V_B =$ Block speed (knots)

$$\frac{1}{E} = \frac{C}{PV_B}$$

Appendix B
SUPPLEMENTARY AERODYNAMICS DATA

In this appendix some of the more important aerodynamic aspects of generalized helicopter design analysis are discussed. This discussion leads to the development of the *aerodynamic required* R_F equation which is necessary to the graphical solution for minimum gross weight, as outlined in Chapter IV and treated in greater detail in Appendix E. The parameter R_F, defined as the ratio of fuel weight to gross weight, is aerodynamically dependent on the cruising power required by the lifting rotor(s), (and tail rotor for single rotor types), the specified range or radius of action and the fuel consumption characteristics of the particular engine type under consideration.

The power required by a helicopter, and the efficiency with which the power is utilized, is dependent upon forward speed, *rotor disk loading w, blade loading w/σ* (σ = the rotor solidity), *blade section drag and lift coefficient* characteristics, *rotor tip speed V_T*, *equivalent parasite drag area A_π*, of the fuselage, empennage, and landing gear, *air density ρ*, and the various losses associated with gearing, tail rotor and accessory drives. Within the flight regime in which stall and compressibility effects are nonexistent, this total power required can be accurately predicted by any one of several established methods. The method used for this study is presented in a separate report.[1] The basic power required formulae arising therefrom are presented as follows.

$$\frac{rhp}{W} = \frac{.031}{\sqrt{\rho/\rho_0}} \frac{\sqrt{w}}{B} K_u$$

$$+ \left[.54\,(\rho/\rho_0)\,\frac{\delta_0 V_T^3}{w/\sigma} + \frac{3.44}{(\rho/\rho_0)} \frac{\delta_2}{a^2} \frac{(w/\sigma)}{V_T} \right] K_\mu$$

$$+ 10.45\,(\rho/\rho_0) \left(\frac{V}{100}\right)^3 \frac{A_\pi}{W}$$

where B = rotor tip loss factor, assumed as .96 for this study

ρ/ρ_0 = air density ratio, altitude to standard sea level

K_u = induced velocity correction factor for forward flight

K_μ = profile power dissymetry correction factor in forward flight

δ_0 and δ_2 = terms in blade section profile drag coefficient expression, $c_d = \delta_0 + \delta_2 \alpha_r^2$

α_r = blade section angle of attack in radians

a = blade section lift curve slope

V = airspeed in knots

and w, w/σ, V_T, A_π are as defined in the preceding paragraph.

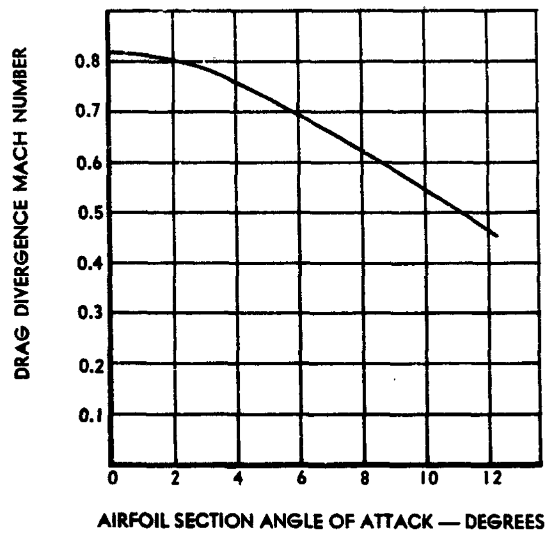
The correction factors K_u and K_μ are explained and presented in chart form in the previously referenced *Transport Helicopter Design Analysis Methods* report. The first term in the above equation is the *rotor induced power*, the second term is the *rotor*

[1] *Transport Helicopter Design Analysis Methods*. HH Report 473.6 30 November, 1955.

profile power and the third term is the *parasite power.*

This power required equation holds only within the flight regime in which stall and compressibility are non-existent. The definition of the *boundaries* of this regime constituted a major effort within the aerodynamic investigations of this study.

The phenomenon of rotor retreating blade tip stall has been adequately investigated by analysis and flight test, and the published results indicate, for most commonly used helicopter rotor blade airfoils (such as the NACA 0012 or 0015) that stall occurs at section angles of attack in the order of 12°, corresponding to section lift coefficients in the order of 1.2. These established values were considered to be adequately representative for the purposes of this study. Rotor compressibility drag rise phenomena are, on the other hand, not so well documented, quantitatively. Various investigators have at times used the limit of *theoretical critical Mach number*, as low as .6 or .7, for most airfoils. It has been demonstrated recently however, by NACA tests (corroborated by high speed flight tests of other helicopter manufacturers, and by propeller tests) that no significant total drag rise occurs below Mach numbers in the order of .75 or .8 at zero section angle of attack, and that Mach numbers approaching .9 or 1.0 at the advancing blade tip *may* be tolerable. Lacking specific proof of the latter hypothesis, the drag divergence Mach number curve shown in Figure B-1 was used as a tentative limit for the general analysis in this study. Using this curve, and a stall limit of $c_l = 1.2$ for the retreating blade tip, existing methods were used to develop the operating limits chart shown in Figure B-2.

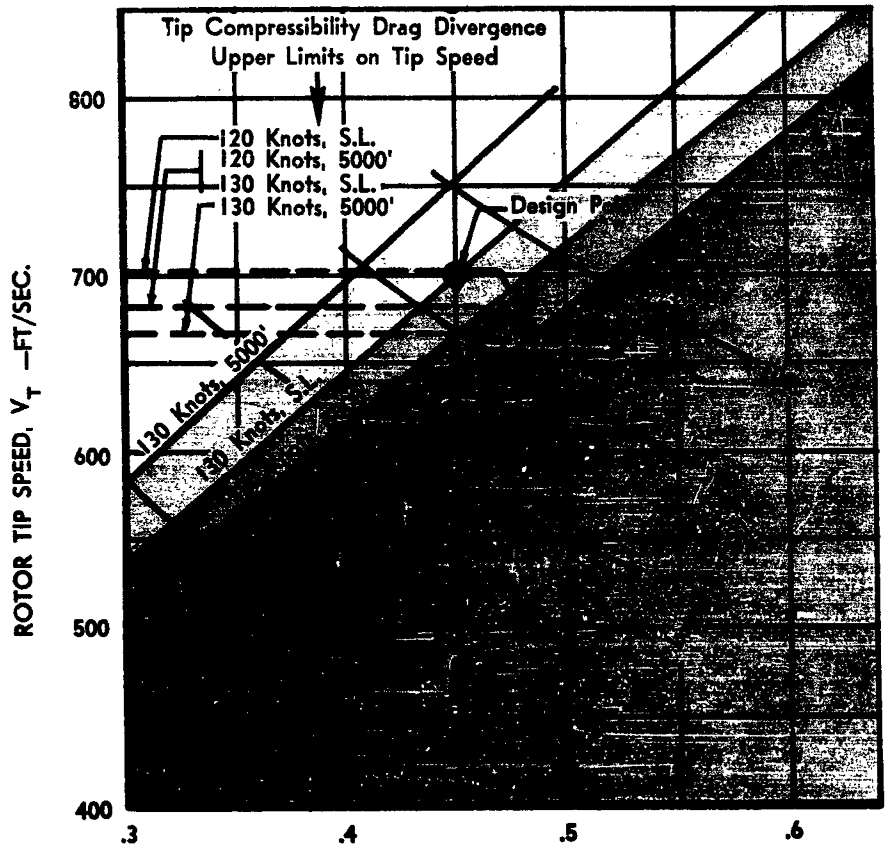


B-1 ASSUMED VARIATION OF BLADE AIRFOIL DRAG DIVERGENCE MACH NUMBER WITH SECTION ANGLE OF ATTACK

This chart incorporates ***airspeed, tip speed, blade loading*** and ***sea level mean blade lift coefficient*** ($C_{L_{r\,sl}} = 6\,(w/\sigma)/\,\rho_o V_T^2$) all into one graphical picture with the operable areas at sea level and 5000 ft. altitude defined by superimposed compressibility and stall limits. Implicit in this chart is the assumption of ***optimum blade twist*** in the order of 6° to 8°, to keep the advancing blade tip at near-zero angle of attack at maximum speed.

For the general study it was tentatively assumed that all helicopters should be capable of at least 120 knots airspeed at sea level. This high speed requirement was selected as representative of current ***best operational*** state of the art. As may be seen from Figure B-2, this requirement imposes a fixed upper tip speed limit of 700 ft/sec, and a lower tip speed limit which varies from 480 ft/sec at the lowest C_{L_r} shown up to 670 ft/sec at

APPENDIX B—SUPPLEMENTARY AERODYNAMIC DATA



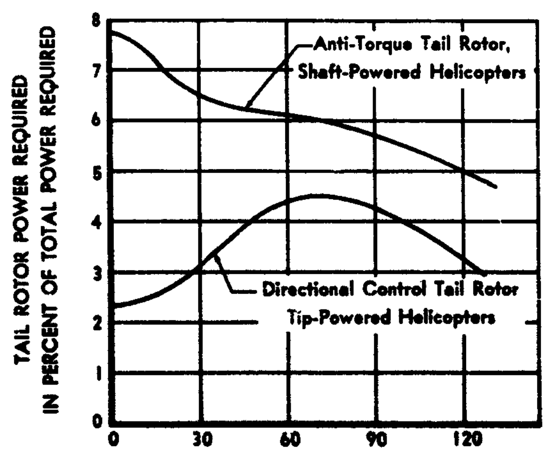
B-2 TIP STALL AND COMPRESSIBILITY LIMITS ON TIP SPEED, FORWARD SPEED, LIFT COEFFICIENT AND BLADE LOADING AT SELECTED ALTITUDES

C_{Lr} = .48. Although pure aerodynamic considerations favor the highest possible C_{Lr} up to the stall limit, a small margin should be allowed for overload and emergency high speed operation above redline limits. For these reasons C_{Lr} at sea level was fixed at .45, slightly less than the maximum value of .48, for all helicopters in the general study. Rotor and transmission weight analyses, which are discussed in Appendix D, indicated that the high tip speed should be selected, since the attendant reduction in rotor weight (due to centrifugal blade bending relief) and in transmission weight (due to decreased torque transmitted) more than offsets the slight power penalty. Thus the design point as shown on Figure B-2 was located at V_T = 700 ft/sec, C_{Lr} @ sea level = .45 and a corresponding blade loading of 87.3 lbs/ft^2. At 5000 ft. cruise altitude, maximum cruise speed corresponding to these assumptions was reduced to 111 knots, due to compressibility limits.

The previously mentioned speculation that tip Mach numbers up to .9 or 1.0 *may* be tolerable, provided sufficient power is available, is based upon the supposition that compressibility drag divergence on the advancing blade tip creates no significant vibratory problem, but only a rise in power required. Recognizing this possibility, a special study of cruise speeds up to 130 knots at 5000 ft was made. For this special case, the stall limits at 130 knots and 5000 ft. shown in Figure B-2 dictated a reduction of C_{L_r} to .39, corresponding to a blade loading of 73.7 lbs/ft^2 at 700 ft/sec tip speed. The increased power required due to the drag rise was, for this case, estimated by a recently developed approximate method.[1]

The tail rotor power for single main rotor helicopters was calculated rigorously for only a few representative main rotor disk loadings and altitudes and converted to a non-dimensional percentage of total power required. The basic power required equation is essentially the same as for the main rotor, except that tail rotor thrust and hence disk loading varies in accordance with the main rotor torque. For the tip powered helicopters, the tail rotor is present only for directional control and therefore in straight and level flight its disk loading is essentially zero which means that only profile power must be supplied. Figure D-1 in a subsequent appendix illustrates the dimensional relationships which were standardized for all single rotor helicopters considered. For the single rotor geared drive types, the anti-torque tail rotor dimensional relationships are representative of those for current large helicopters, within ± 5%. The dimensions for the single rotor tip drive tail rotors were established as being optimum compromises between low power, low diameter, for long tail booms and slightly higher power and higher diameter for short tail booms. The analysis from which the dimensions were determined was based on hovering directional control specifications given in Military Specification MIL-H-8501. Figure B-3 shows the standard variations of percent tail rotor power with forward speed for the two tail rotor types.

Figure D-1 in Appendix D illustrates the assumed dimensional relationships for the tandem rotor helicopters. The 60% overlap was selected on the basis of studies which indicated that gross weight decreases with increasing overlap, the primary influence being the fuselage and drive shaft weight decreases, offsetting the small power increases due to



B-3 VARIATION OF PERCENTAGE TAIL ROTOR POWER WITH AIRSPEED

[1] Kenneth B. Amer. *Effect of Blade Stalling and Drag Divergence on Power Required by a Helicopter Rotor at High Forward Speed.* Proceedings of the Eleventh Annual Forum, American Helicopter Society, April, 1955.

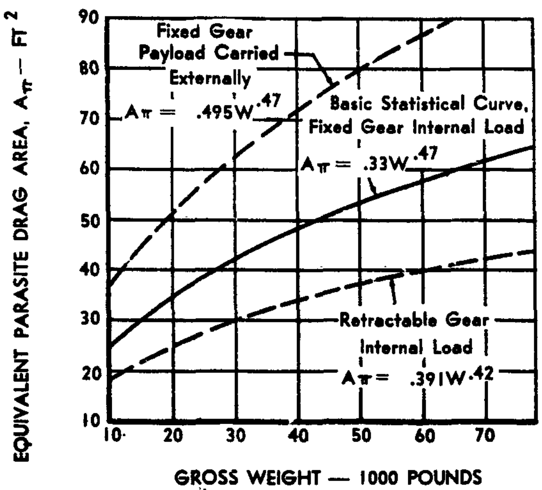
APPENDIX B—SUPPLEMENTARY AERODYNAMIC DATA

aerodynamic interference effects between rotors. A vertical gap of .1 times the blade radius was quite arbitrarily assumed for the tandems (rear rotor higher than the front rotor). With this small gap, the rotors are essentially intermeshing, hence there was a certain maximum overlap, about 80%, beyond which blade lag motions would be a mechanical interference problem. However, since 80% overlap is considerably higher than the statistical average for the majority of large tandem helicopters, and since the weight savings are small as overlap is increased beyond 50 or 60%, the latter value was used. In the calculation of tandem rotor power required, the only difference from the single rotor type calculations lies in the omission of the tail rotor, and in a somewhat higher induced power. This latter effect is accounted for in the power required equation by the use of a higher induced velocity correction factor, K_u, the derivations and curves for which are presented in the previously referenced *Design Analysis Methods* report.

The assumed variations of *equivalent flat plate parasite* area, A , are plotted in Figure B-4 for

1. *Basic helicopters* with fixed landing gear, and payload carried internally
2. Helicopters with *retractable landing gear*, and payload carried internally
3. Helicopters with *payload carried externally*, and fixed landing gear.

The first curve is based on statistics[1], and the latter two were developed by generalized analyses of landing gear drag per pound gross weight, and drag of high density loads carried externally by a cargo sling or net. The first



B-4 VARIATION OF EQUIVALENT PARASITE DRAG AREA WITH GROSS WEIGHT

curve shows good agreement, above 5000 lbs. gross weight, with drag estimates for a wide variety of helicopter types and sizes. It was found that, within the required accuracy of this stu'y, no significant difference in parasite drag for single rotor as opposed to tandem rotor helicopters could be ascribed, albeit the fuselages for the two types differ markedly in appearance.

The total brake horsepower required per pound gross weight was calculated from the previously discussed *rotor* power required, by the following equation

$$\frac{Bhp}{W} = \frac{1}{\eta}\,\frac{rhp}{W}$$

where η is the propulsive efficiency, including tail rotor power loss, where present, and the gearing losses.

It should be mentioned that this power required equation does not include a correction factor for *vertical drag*, which has manifested itself as a significant airload characteristic of *stub-wing helicopters* and *convertiplanes*. This factor, arising from the impingement of high

[1] *Helicopter Propulsion System Study*; Thermal Research and Engineering Corporation, Conshohocken, Pa.; September, 1952.

rotor downwash velocity on wing, tail surfaces, and fuselage, was neglected on the basis that:

1. No helicopter in this study incorporated stub-wings.
2. For twin reciprocating powered versions, the design studies indicated that the engine installation could be so designed as to avoid a *flat plate* blocking effect of a significantly large area.

and 3. Fuselage cross sections were assumed to be sufficiently rounded on top to yield a low vertical drag coefficient.

Given the *power loading* (a function of disk loading and hover ceiling only, as discussed in Chapter IV), and the cruise power at a series of forward speeds up to the previously discussed rotor-limited speeds, the determination of fuel SFC and aerodynamic R_F was a relatively straightforward procedure, outlined in sequence below:

1. Power setting in percent of sea level normal rated power was calculated from the power required at each speed, and the power loading
2. Fuel SFC was read from the charts, Figures C-3 and C-4, in the following appendix.
3. Fuel rate per pound gross weight per nautical mile, defined as dR_F/dR, was calculated from

$$\frac{dR_F}{dR} = \frac{(SFC)}{V} \frac{Bhp}{W}$$

where V is airspeed in knots.

4. Minimum dR_F/dR and corresponding cruise speed was determined from curves of dR_F/dR versus V.
5. In accordance with the standard mission flight plan discussed and illustrated in Chapter 5, the aerodynamic required R_F for each specified radius of action was calculated as the sum of the ΔR_F in climb, ΔR_F in cruise, ΔR_F for start and warm-up and ΔR_F for reserve fuel (10%) as follows:

$$\text{Total fuel} = \text{climb fuel} + \text{cruise fuel}$$

$$R_F = \Delta R_{F_1} + \Delta R_{F_2}$$

$$\text{start fuel} + 10\% \text{ reserve}$$

$$+ \Delta R_{F_3} + 0.1\, R_F$$

or

$$R_F = 1.11\ (\Delta R_{F_1} + \Delta R_{F_2} + \Delta R_{F_3})$$

where:

climb fuel

$$\Delta R_{F_1} = 2\left(\frac{h_1 - h_0}{R/C}\right)\left(\frac{V_{C1}}{60}\right)\left(\frac{dR_F}{dR}\right)_0$$

(2 equal climbing legs outbound and inbound.)

cruise fuel

$$\Delta R_{F_2} = 2\left(R - \frac{h_1 - h_0}{R/C}\ \frac{V_{C1}}{60}\right)\left(\frac{dR_F}{dR}\right)_{min.}$$

(2 equal cruise legs, outbound and inbound)

start fuel

$$\Delta R_{F_3} = 2\,\frac{t_s}{60}\left(\frac{dR_F}{dt}\right)_0$$

(2 engine starts, outbound and inbound.)

h_1 = cruise altitude—5000 ft.
h_0 = base altitude—4000 ft.
R/C = average maximum rate of climb, ft/min
V_{C_1} = airspeed for maximum rate of climb, knots

APPENDIX B—SUPPLEMENTARY AERODYNAMIC DATA

$\left(\frac{dR_F}{dR}\right)_0$ = full throttle fuel rate per pound gross weight per nautical mile

R = Radius of action

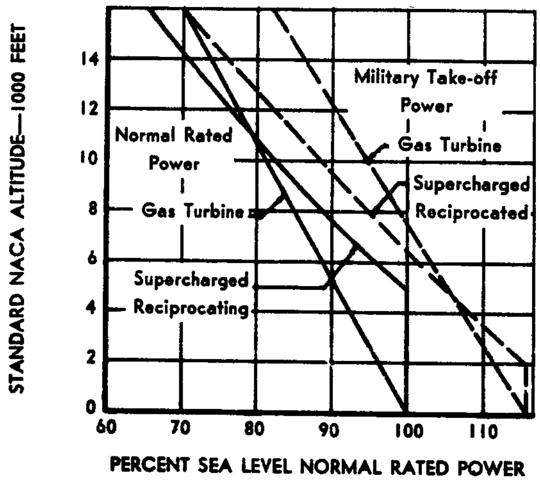
$\left(\frac{dR_F}{dR}\right)_{min}$ = cruise fuel rate per pound gross weight per nautical mile, at cruise altitude h_1

$\left(\frac{dR_F}{dt}\right)_0$ = full throttle fuel rate per pound gross weight per hour

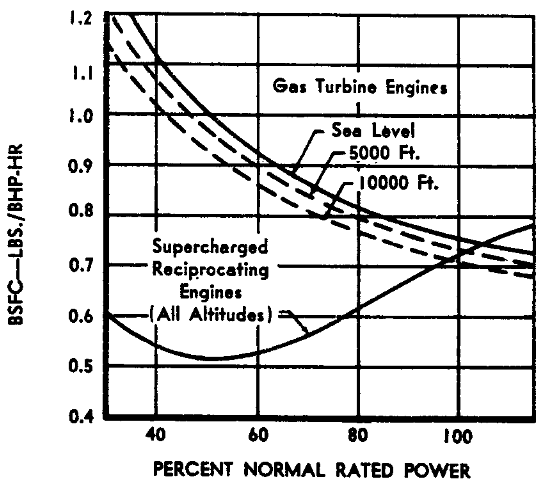
t_s = time for one start and warmup

The application of the aerodynamic R_F in the graphical analysis to determine minimum gross weight is discussed in Appendix E.

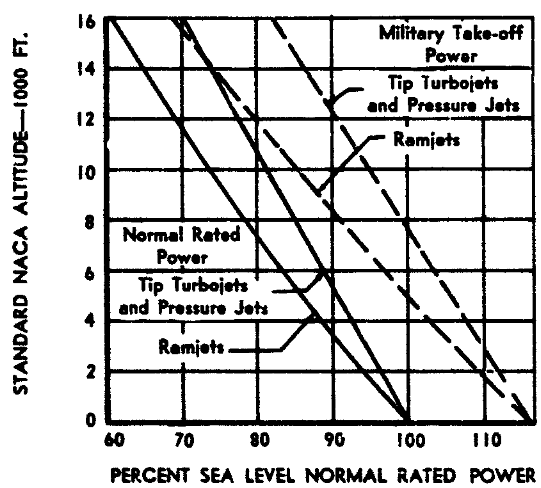
Appendix C
SUPPLEMENTARY POWERPLANT DATA



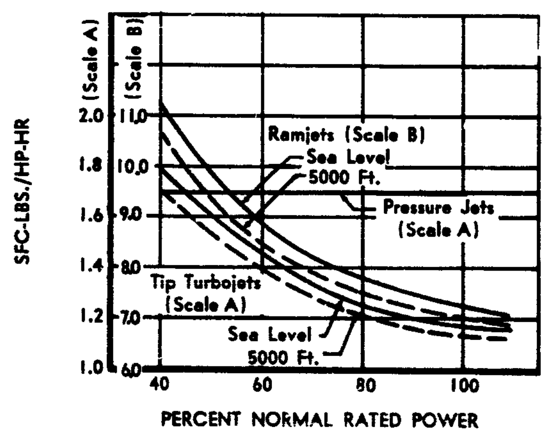
C-1 GENERALIZED POWER VARIATION WITH ALTITUDE. GEARED ENGINES



C-3 GENERALIZED VARIATION OF BSFC WITH PERCENT NORMAL RATED POWER FOR GEARED ENGINES



C-2 GENERALIZED POWER VARIATION WITH ALTITUDE FOR TIP DRIVE ENGINES



C-4 GENERALIZED VARIATION OF SFC WITH PERCENT NORMAL RATED POWER FOR TIP DRIVE ENGINES

Appendix D
SUPPLEMENTARY WEIGHT DATA

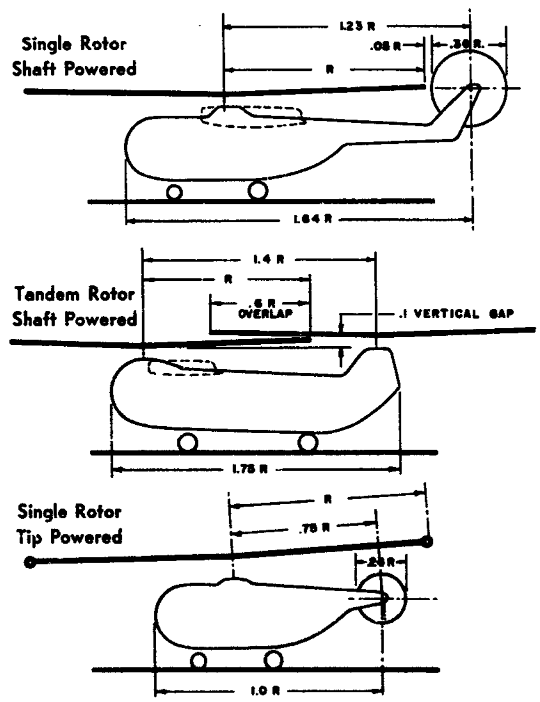
1. Introduction

The objective of the weight analysis was primarily to determine the trends of empty weight with the major design parameters of disk loading, tip speed, hover ceiling, fuselage length and rotor overlap (in the case of tandem helicopters). Having the pertinent empty weight trends, the next step was to obtain the allowable fuel weight for a given design payload requirement and to combine the *allowable* fuel weight function with the *required* fuel weight function to obtain the simultaneous solution of aerodynamic and weight equations described in Appendix E.

The purpose of this appendix is to outline the methods and assumptions used to obtain empty weight and fuel weight as functions of the design parameters. The methods are applicable to transport helicopters only, and a more general treatment of helicopter weight data and analysis is presented in the report, *Design Analysis Methods*[1]. General comments on the validity of the statistical data used herein and the probable accuracy of the results are contained in the report mentioned above and in Chapter IV of this report.

2. Approach to the Problem

The first step in the analysis was to assign a fixed value to each major helicopter dimension that would influence the weight analysis. Both the aerodynamic and weight analyses were thereafter based upon these standard dimensions, which are shown in Figure D-1 for the three main configurations. For single rotor shaft powered helicopters, the dimensions are typical of current design practice, and the overall body length of 1.64 x rotor radius is the statistical average shown by the Thermal Research and Engr. Corp. report[2].



D-1 GENERALIZED DIMENSIONS FOR VARIOUS DESIGN CONFIGURATIONS

[1] H.H. Report 473.6; *Transport Helicopter Design Analysis Methods*; 30 November 1955.

[2] *Helicopter Propulsion Systems Study*; USAF Contract 33(038)-22185; Thermal Research and Engineering Corp.; September, 1952.

Since the overall fuselage length did not directly enter into the aerodynamic analysis, no attempt was made to optimize the dimension other than to ensure that decreasing rotor radius would not limit the allowable fuselage cargo space. The method used to accomplish this is treated in more detail in part 3 of this Appendix.

Tandem rotor fuselage length, being a function of rotor overlap, was the subject of a separate analysis, in an attempt to determine an optimum. It was found that for any given design payload and radius of action, the resulting design gross weight decreased with increasing overlap. Increasing overlap decreases body length and hence body weight (statistically), while at the same time increasing the rotor diameter and hence weight for a given effective disk loading. These opposite effects combined to yield lower empty weight for higher values of overlap, i.e., the rate of decrease of body weight with overlap was in all cases greater than the rate of increase in rotor weight. Thus the value of body length (shaft to shaft) and rotor overlap were established by the criterion of maximum cargo dimensions as the lower limit. The body overhang beyond the rotor masts and the vertical displacement of the rotors were assigned values consistent with current design practice.

For the single rotor tip powered configurations, the body length was fixed at 1.0 x rotor radius in accordance with cargo dimensional requirements. Tail rotor radius was determined by the criterion set forth in MIL-H-8501[1] which specifies a minimum yaw requirement. The following three factors involved in tail rotor design were therefore made consistent with the specified requirement: adequate main rotor-tail rotor clearance, ground clearance, tail rotor radius and power requirements. From these considerations, tail rotor radius was fixed at .12 x main rotor radius. The resulting tail rotor power required is given in the aerodynamic analysis, Appendix B.

The second step of the analysis was to formulate an equation which would express helicopter empty weight in terms of the design parameters. The Thermal Research report presented a source of statistical data for helicopter component weights and analytical expressions for the weight relationships. For a given rotor configuration, power plant type, and number of installed engines, it was possible therefore to write an analytical expression for the helicopter empty weight in terms of disk loading (w), tip speed (V_T), gross weight(W), and take-off power loading (l_p). By the analysis discussed in Chapter IV, it was concluded that higher tip speeds would result in higher rotor rpm and hence lower transmission torque for a given power loading. Aerodynamic consideration of advancing blade drag divergence limited the tip speed to 700 ft/sec which was therefore selected for all configurations in the study. The final empty weight equation therefore reduced to a function of gross weight, disk loading, and take-off power loading (defined for the study as gross weight divided by the take-off engine horsepower).

3. Major Component Weights

Rotor group weight trends from the statistical data were based on two primary types: rotors having three articulated blades and rotors with two rigidly interconnected (teetering) blades. The blade weight and hub and hinge weight for these two types, when combined, showed a small weight advantage for

[1] MIL-H—8501; Requirements for Helicopter Flying Qualities; 5 November 1952.

APPENDIX D—SUPPLEMENTARY WEIGHT DATA

the articulated rotor blade system, hence this system was used as a basis for rotor weights for all shaft driven helicopters in the study.

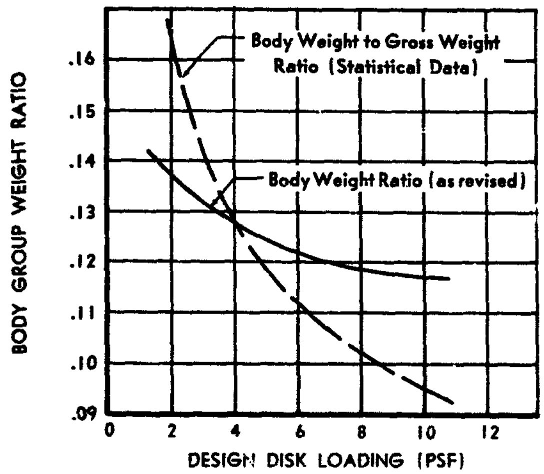
Rotor group weights for tip powered helicopters were essentially based on two bladed rotors from the statistical data.

The rotor group weight expressions in general were functions of disk loading, tip speed and gross weight, however, no significant correlation was found with solidity. Recent analysis[1] has indicated a variation of blade weight with solidity such that blade weight was proportional to the square root of solidity. In this study, since blade loading (w/σ) was fixed and disk loading a variable, solidity (σ) varied in the extreme from .023 to .115. A possible error of approximately 2% in gross weight has been incurred by neglecting the variation of rotor weight with solidity, based on the assumption that the statistical data represents helicopters with a mean solidity of .045.

The tail rotor weight expression was parametrically similar to the main rotor weight function. Tail rotor tip speed was also taken as 700 ft/sec., and tail rotor disk loading for shaft drive helicopters was determined by the thrust required to balance engine torque at engine take-off power. As noted in part 2 of this Appendix, tail rotor thrust and design disk loading for tip powered helicopters were determined from the maneuvering requirements of MIL-H-8501.

Fuselage weight, from the statistical data, was found to be a function of helicopter gross weight and overall fuselage length. From the relationships shown in Figure D-1, the weight may also be expressed in terms of gross weight and disk loading. It followed that increasing disk loadings decreased fuselage weight. After preliminary analysis of the single rotor helicopters, it was found necessary to revise the weight expression to show less effect of variation in disk loading on body weight. Specifically, as disk loading increased, body length was found to decrease to the point where a fixed cargo compartment length was a much larger percentage of overall length than corresponding single rotor machines represented by the statistical data. Since the cargo compartment or pod is structurally heavier per unit length than tail boom, and since decreasing body length for a given payload merely results in reducing tail boom length, the body weight expression was revised as shown by the comparison in Figure D-2 to obtain a lower slope with disk loading. The statistical data were assumed to be representative of helicopters with disk loadings of four, hence the intersection of the two curves at that point.

Power plant weight, primarily engine weight, also included the weight of items which would



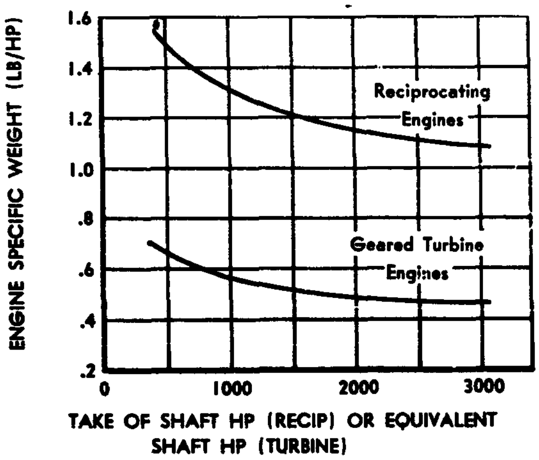
D-2 BODY WEIGHT TO GROSS WEIGHT RATIO VS. DESIGN DISK LOADING
- Single rotor shaft drive helicopters
- 10,000 lb. gross weight

[1] *Application of Statistical Weight Analysis Methods to Helicopter Preliminary Design*; George R. Holzmeier; presented at the SAWE Conference May 1955.

normally be a function of the installed engine power and therefore a function of engine weight. Engine accessories and controls, cooling and lubrication systems, oil, oil tanks, starting systems, and fuel and oil systems are included in this group since their weights are all functions of installed power.

The basic weight equation for each of the four shaft drive configurations were written to allow for from one to four installed engines. A single engine installation however, was obviously the most efficient weightwise, since for any given total installed power the individual engine power varied inversely with the number of engines; and as Figure D-3 shows, the specific weight (lb/HP) increases at low power ratings. Therefore, the primary results of the study were based on helicopters with twin engine installations as a compromise between weight and cost penalties on the one hand incurred by multiple engines, and by increased flight reliability on the other.

Ramjet engine weights were based partially on the data of the Thermal Research report with some modifications derived from studies conducted at Hiller[1]. These studies were based upon strength-weight analyses which considered the detailed effects of centrifugal "g" field, engine diameter and temperature on engine weight. Combination of these data resulted in a specific weight (lb/lb of thrust) somewhat more conservative than the statistical data. Considering however, the percentage of empty weight which these engines comprise, it was felt that negligible error was incurred by these conservative specific weights. Tip turbojet engine weights have been discussed previously in Chapter IV, Part C. Pressure jet power plants, for lack of statistical weight data on the basic turbine and compressor units, were based on an analysis of projected capabilities of this power plant type. This analysis is appended to the referenced *Design Analysis Methods* report. The analysis assumed an average power plant weight of 0.5 lbs/shaft HP for the geared turbine and secondary compressor units. Tip burner weights and additional ducting were assumed to be included in the rotor weight for these



D-3 POWERPLANT SPECIFIC WEIGHT VS. RATED HP.

comparatively low weight items.

Transmissions and drives weights for each component were obtained from the statistical data as a function of the maximum torque transmitted.

For single rotor helicopters, the transmissions and drives were assumed to consist of rotor mast, main and tail rotor transmissions, and tail rotor drive shaft. From the aerodynamic analyses the installed power was obtained for any given hover ceiling and disk loading. Thus it was possible to express the torque and hence weight of each component in terms of the power loading, tip speed, disk loading and gross weight.

[1] H.H. Report 250.7; *Study—Optimum Ramjet Engine Weight*; October 1952.

APPENDIX D—SUPPLEMENTARY WEIGHT DATA

For all configurations included in the study, the transmissions and drives were "designed," weightwise, to carry the torque corresponding to engine take-off power. Since hover ceilings considered in the study matrix resulted in helicopters with extremely high rates of climb, i.e., considerable excess power was available *at sea level*, it would be possible to "derate" the engine at sea level. This would result in lower design torque for transmissions and drives and would have resulted in some decrease in empty weights. By this procedure, adequate sea level rates of climb would be assured while obtaining the weight advantage of lower "design" torque loadings on the transmissions and drives.

Tandem rotor helicopter transmissions and drives were assumed to consist of two rotor masts, two main transmissions, a rotor interconnect shaft, and an intermediate gear box. Main transmissions were assumed to be planetary reduction units and both main transmissions and rotor masts were "designed" to carry 60% of the torque developed at take-off engine power.

For the shaft powered helicopters considered in the study, the effect of multi-engine installations was to increase the weight of transmissions and drives components. The available statistical weight data did not include sufficient detail to evaluate this increase, and the weight increase of multi-engine installations of transmissions and drives was handled in the following manner. The group weight was increased by the weight of additional engine-transmission shafting required; and transmission weight, due to the additional bearings and gear sets required, was increased by the ratio shown in a recent transmission study[1]. This resulted in a 16% increase in main transmission weight for single rotor helicopters and an 11% increase in intermediate gear box weights for tandem rotor helicopters.

For the tip powered helicopters, transmission and drives weight was found to be considerably smaller since the power transmitted by the main transmission served only to drive the tail rotor and accessories. "Design" torque for these items was therefore determined primarily by tail rotor power requirements. Rotor mast weight was conservatively assumed to be the same as the mast weight for a shaft powered helicopter of equivalent power loading and disk loading.

Normal definition of *fixed equipment weight* includes flight controls, hydraulic and electrical systems, instruments, personnel and crew accommodations, heating and air conditioning and emergency provisions. From the statistical data these individual weight items were analytically expressed as functions of gross weight. Two additional weight items, landing gear, and stabilizer, were also analytical functions of gross weight, hence the total "fixed equipment" group consisted of stabilizer, landing gear and fixed equipment (in the normal sense) weights. The available data showed weight trends for *each* item in the fixed equipment category, however it was felt that since the helicopters represented by the statistical data were designed for many varied missions and therefore had varied types of equipment, a better correlation of the data could be obtained. With the cooperation of the Thermal Research and Engineering Corp., the original data was retabulated and the *total* fixed equipment weight for each of the helicopters comprising the data was plotted vs. the design gross weight. As anticipated, the

[1] *The Price of Helicopter Transmission Service Life*; R. M. Carlson and F. D. Schnebly; presented at the Army Transportation Corps Conference and Symposium on Spare and Replacement Parts; May, 1954.

new correlation showed a trend of lower slope than that previously found. An additional, but equally important aspect of the new correlation was that there appeared to be no significant difference between single and tandem rotor fixed equipment weights. This revised trend was therefore used throughout the study.

The one item omitted from the final trend of fixed equipment weights was communications equipment. The following units were considered to be requisite for the transport mission and the corresponding total weight was included for each configuration:

VHF Command Set ARC-3	94.0 lbs.
Auto. Radio Compass ARC-7	84.0
Identification Equipment (IFF)	53.0
Interphone	31.0
Marker Beacon Equipment	11.0
Total Communications Equipment	273.0 lbs.

4. Payload Weight Equation

Derivation of the weight equation. As has been discussed in Chapter IV and in Appendix E, the final objective of the weight analysis was to express the allowable fuel weight as a function of gross weight, design disk loading and design hover ceiling. Based on the analytical expressions for the items of empty weight it was possible to write the allowable fuel weight as a function of the design parameters, thus:

$$W_{FT} + W_F = W - W_P - W_C - W_E$$

where W_{FT} = fuel tank weight
W_F = fuel weight
W = gross weight
W_P = payload
W_C = crew weight, assumed to be 600 lbs.
W_E = empty weight less fuel tank weight

In ratio form, dividing by gross weight:

$$R_{FT} + R_F = 1 - R_P - R_C - \Phi$$

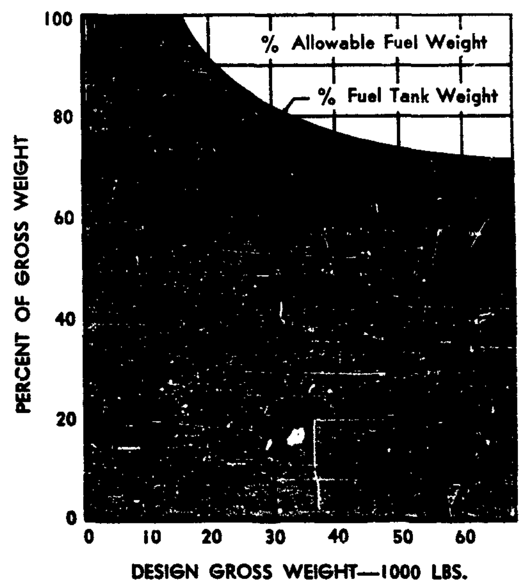
where the symbol Φ has been designated as the ratio of empty weight less fuel tank weight to gross weight. Fuel tank weight was omitted from empty weight since the weight equation is not a function of design radius of action. Therefore the tank weight ratio was linked directly to R_F by assuming a tank weight of 0.5 lb per gallon of fuel. The final weight equation became:

$$R_F = K\,(1 - R_P - R_C - \Phi)$$

where K = the ratio of fuel weight to fuel plus tank weight: 6.0/6.5 = .923 for purposes of this study.

The values of Φ were calculated for the matrix of design disk loading, hover ceiling, rotor and power plant configurations. These values are shown in Figures D-5 through D-25 plotted against gross weight.

Figure D-4 shows graphically the effect of various factors in the weight-payload equation for a *single* typical payload. The vertical scale indicates the cumulative percentage of item weight ratios, thus the manner of variation of the primary weight factors can be readily visualized. The empty weight curve is a typical variation of the term Φ It may be seen that the ratio of payload and crew weight decreases hyperbolically with gross weight since they are fixed weight items. The remaining ***available*** weight ratio must be divided between fuel and fuel tank weight ratios such that at the point of zero R_F (zero radius of action), fuel tank ratio is zero. This ratio then increases with increasing allowable fuel weight ratio. After deduction of tank weight ratio the incremental ratio remaining, R_F, may be equated to the required aerodynamic fuel weight ratio resulting in compatible solutions to both payload and range criteria.



D-4 VARIATION OF EMPTY AND USEFUL LOAD WEIGHTS WITH GROSS WEIGHT

Variation of Empty Weight less Fuel Tank Weight Ratio with Gross Weight. The curves shown in this section formed the basis for the R_F analysis of all configurations included in the study. The characteristic shapes of the trends for each configuration are the result of the interrelationships of many factors.

In general, the effect of increasing hover ceiling for a given configuration, increased the empty weight ratio by virtue of the increased powerplant and mechanical drives weight. Where this weight formed a larger percentage of empty weight, increasing hover ceiling caused a larger increase in empty weight. Thus, reciprocating engine powered helicopters suffered greater increase in empty weight with increasing design hover ceiling than did the geared gas turbine powered machines; whereas the tip powered helicopters suffered negligible empty weight increase with a comparable increase in design hover ceiling.

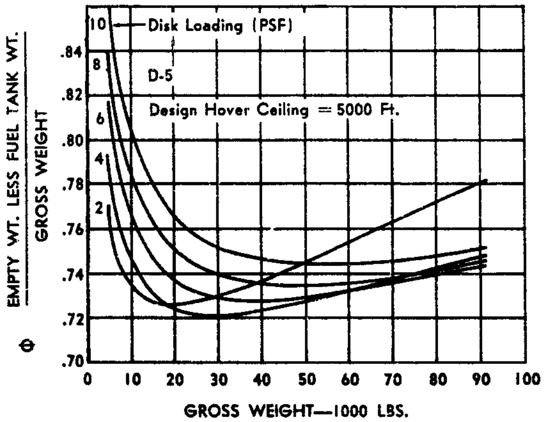
The effect of disk loading on the empty weight ratios is again dependent on the type of rotor and powerplant configurations. Reciprocating powered machines of *low* gross weight showed *increasing* empty weight ratios with increasing disk loading, whereas at *higher* gross weight, the reverse obtained, where increasing disk loading caused *decreasing* empty weight ratio. Physically these phenomena were caused by the relative importance of powerplant weight on the one hand and rotor and body weight on the other, since increasing disk loading has the effect of increasing installed power and, opposing this, decreasing rotors and body weight. At low gross weights the rotors and body group weights are relatively small compared to powerplant weights, therefore the predominant factor in the trend to low empty weight at low disk loadings follows the trend of powerplant weight with disk loading. At higher gross weights however, rotor and body weights become predominant and the variation of empty weight follows this trend, i.e., lower empty weight with higher disk loadings.

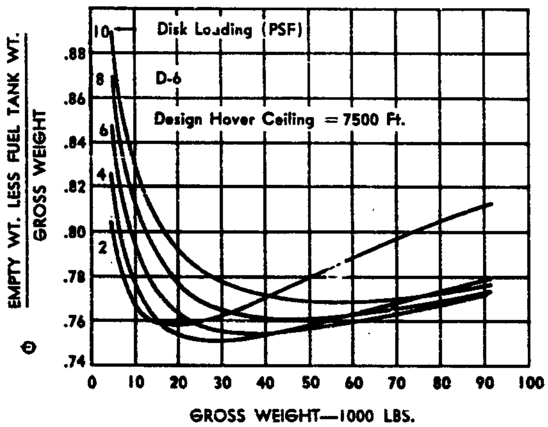
For geared gas turbine or tip powered helicopters the trend was in the direction of decreasing empty weights with increasing disk loading for all gross weights. This trend was attributed to the relatively low weight of geared turbine and tip powerplant types and the relatively high weight of rotor and body groups.

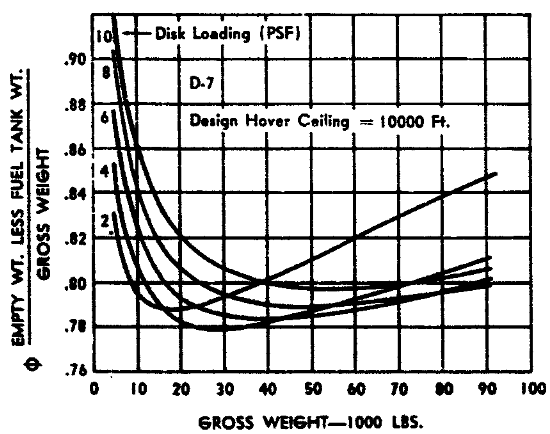
It will also be noted that the slope of the empty weight curves for tandem rotor helicopters is lower than that for the single rotor machines for any given disk loading and powerplant type. This effect is due to the exponential increase of transmissions and rotor weight with gross weight. Although tandem helicopters require duplicate transmissions and rotors, the rate of increase in total weight of these items for the same gross weight and disk loading is in favor of the tandem due to the lower rate of weight increase at the lower radius for rotors and lower "design" torque for transmissions. For this reason, plus the fact that tandem helicopters are more efficient in hover, requiring lower installed power for a given hover ceiling, the level of tandem empty weight ratios were found to be slightly lower than the single rotor helicopters.

APPENDIX D—SUPPLEMENTARY WEIGHT DATA

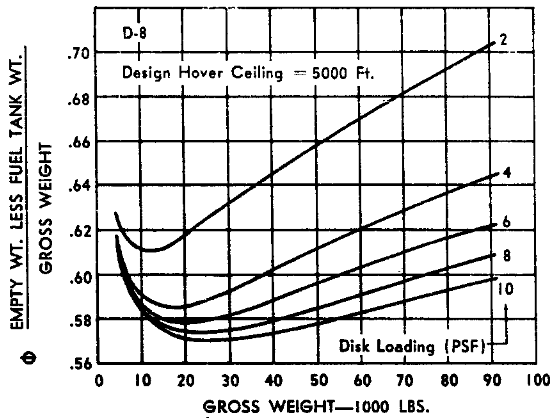
- Single rotor helicopters
- Twin reciprocating engines

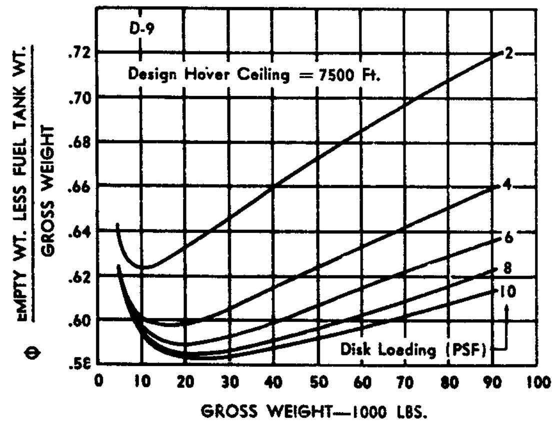


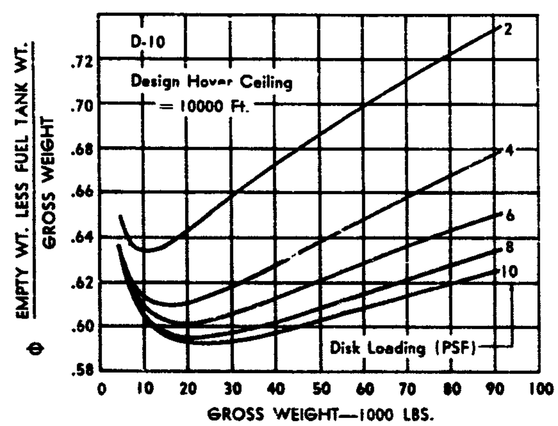




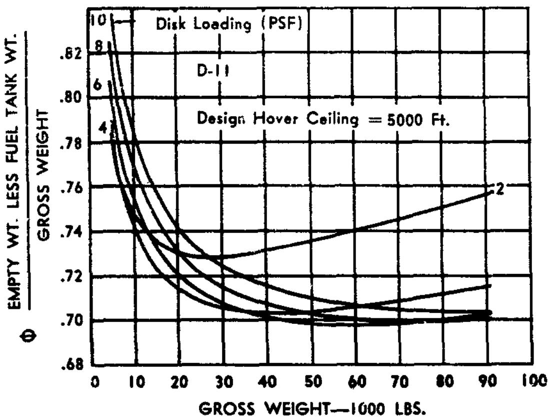
- Single rotor helicopters
- Twin geared turbines

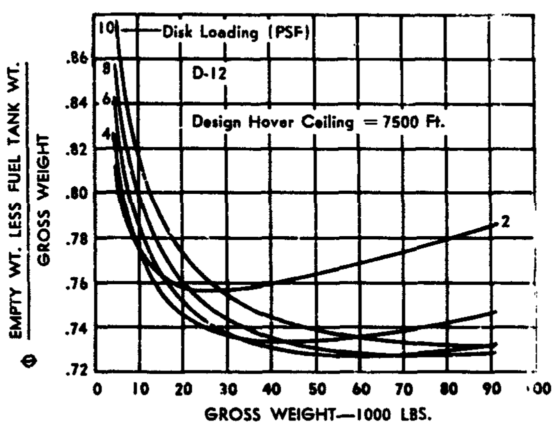


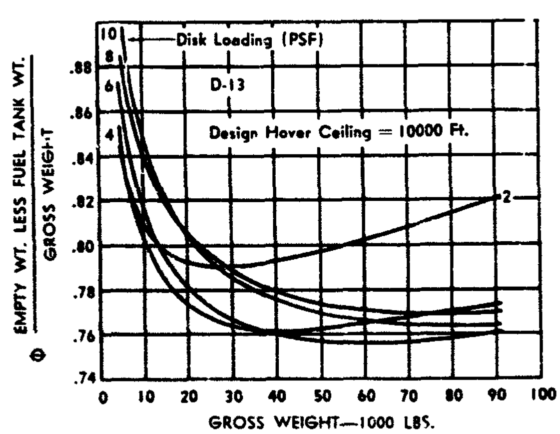




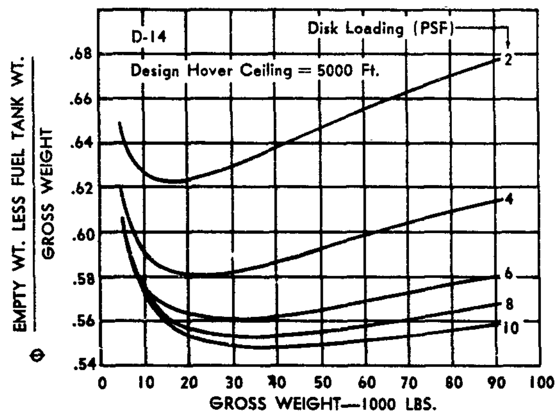
• Tandem rotor helicopters
• Twin reciprocating engines

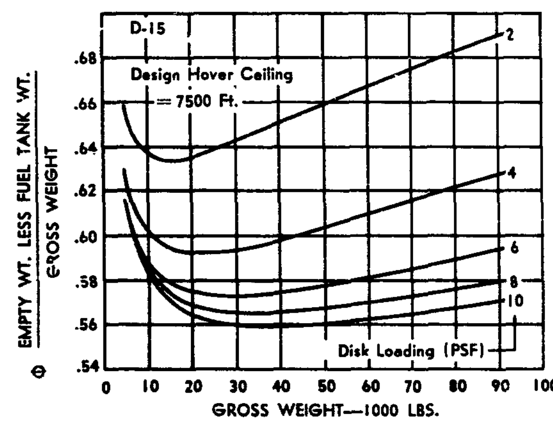


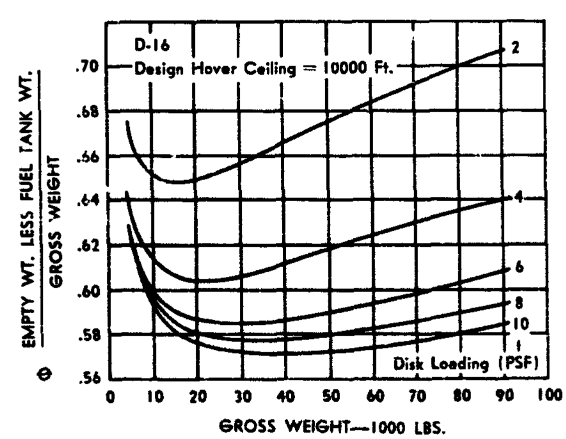




• Tandem rotor helicopters
• Twin geared turbines

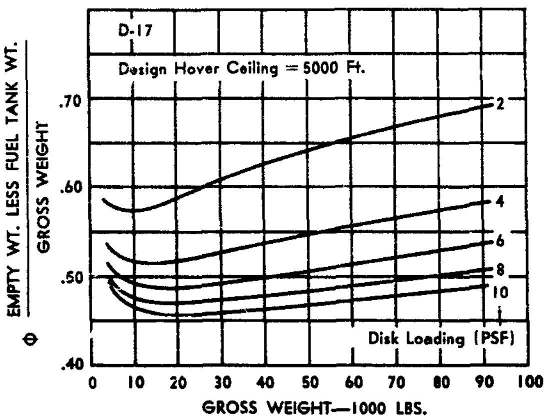


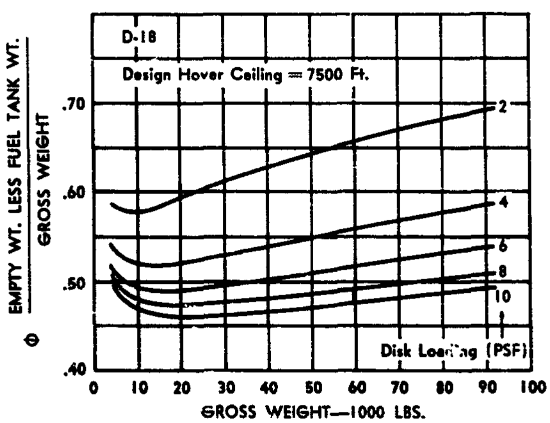


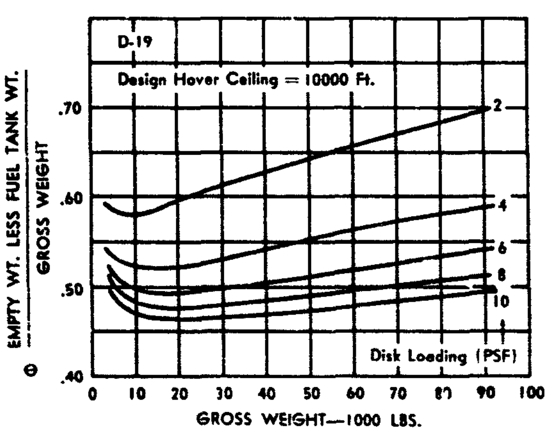


APPENDIX D—SUPPLEMENTARY WEIGHT DATA

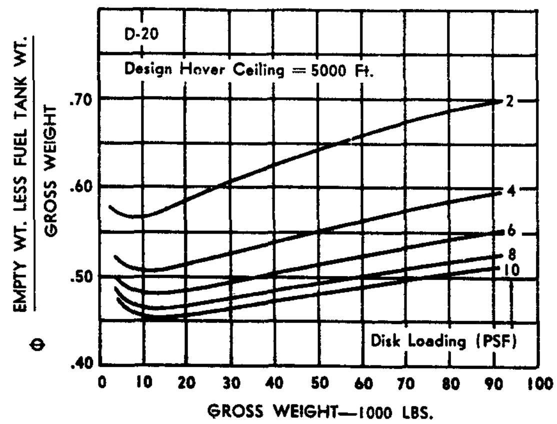
- Single rotor helicopters
- Tip mounted ramjet engines

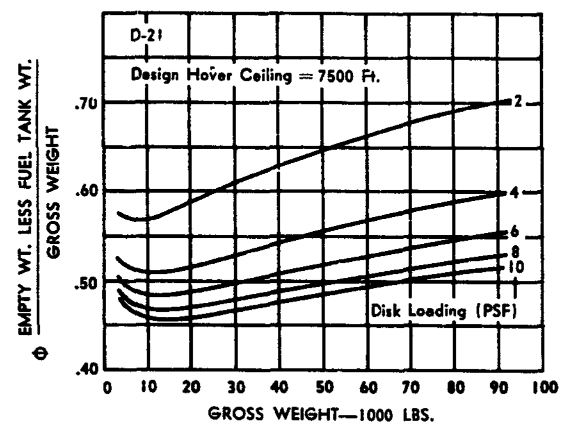


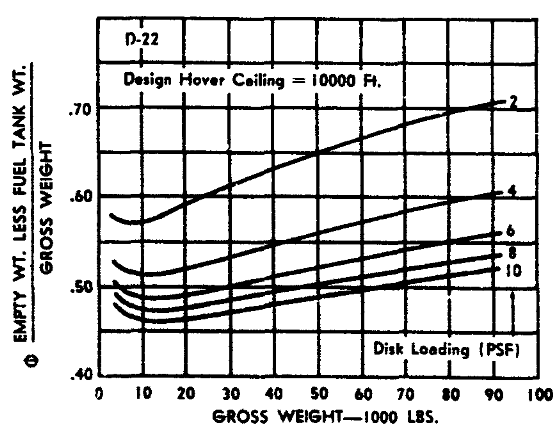




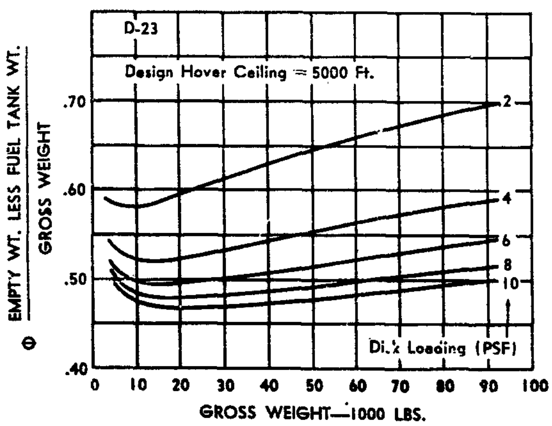
- Single rotor helicopters
- Tip mounted turbojet engines

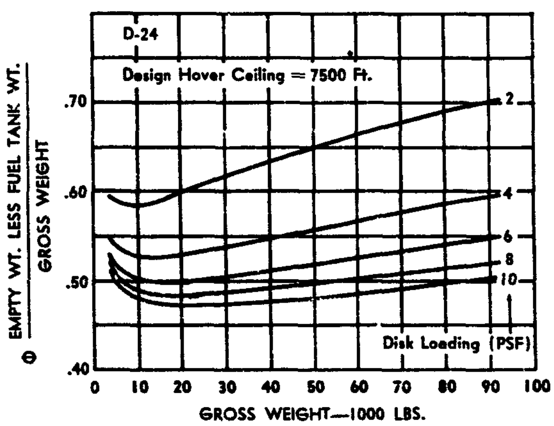


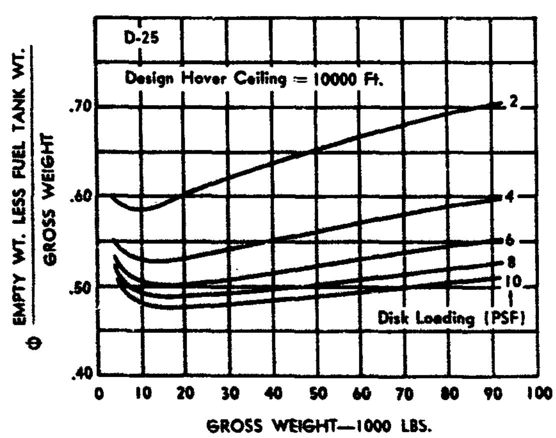




• Single rotor helicopters
• Pressure jet powerplant





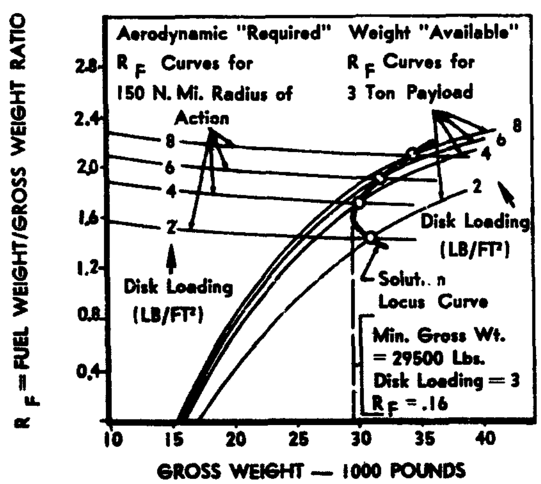


Appendix E
DESIGN PARAMETER SELECTION TECHNIQUE

The design parameter selection technique is essentially a graphical solution method for enforcing compatibility between aerodynamic and weight characteristics through the common link formed by the R_F *ratio*, defined as the ratio of fuel weight to gross weight. In effect, this graphical technique sets the *aerodynamic required* R_F equal to the *weight available* R_F, for each unique combination of disk loading, hover ceiling radius of action, and payload. The formulae and built-in assumptions for the calculation of aerodynamic and weight R_F have been outlined in Appendices B, C, and D. As shown, the *aerodynamic required* R_F is a function, within the assumptions made herein, of disk loading, power loading, gross weight and design radius-of-action, and is independent of payload. Conversely, the *weight available* R_F is a function of disk loading, power loading, gross weight and payload, and is independent of radius of action. Futhermore, since power loading is directly related to and determined by the hover ceiling for a given disk loading, the power loading and hover ceiling are in effect *interchangeable* when speaking of the common functional ingredients affecting the aerodynamic and weight R_F equations.

The first step in the graphical solution method is to plot the aerodynamic and weight R_F curves versus gross weight, for each of an organized family of parameters. Figure E-1 is an example of this type of graph. The aerodynamic curves of R_F are nearly flat, decreasing with increasing gross weight. This trend is due entirelv to the inherent "square-cube" (area-volume) law by which larger machines become progressively "cleaner" aerodynamically, as manifested by the formulae for equivalent parasite drag area presented in Appendix B.

The weight curves of R_F versus gross weight exhibit a marked concavity and tendency to maximize R_F at some point. This trend is primarily a manifestation of increasing percentage weight of the rotor system and transmission and drives system for a given disk loading (or hover ceiling), as the size of the machine increases. This phenomenon is of course quantitatively dependent upon the "state-of-the-art."

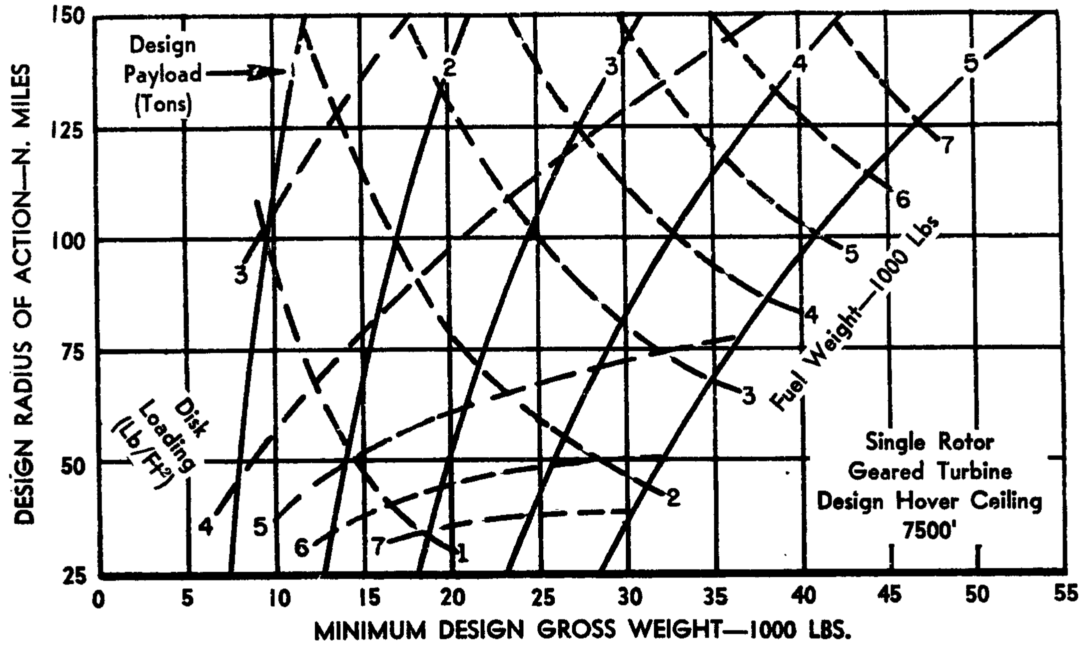


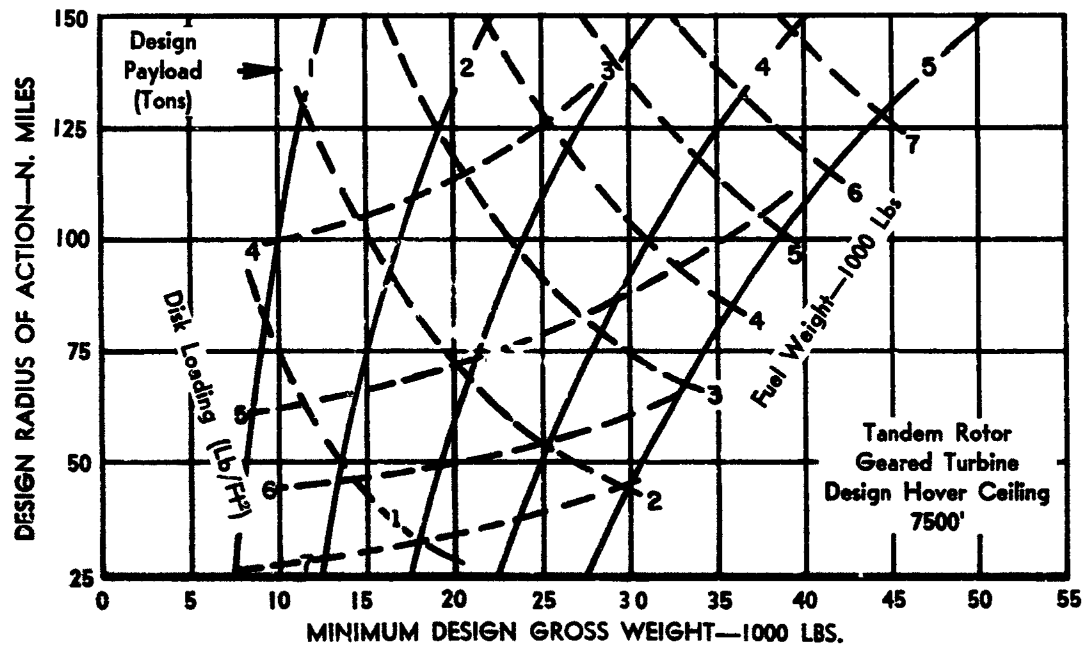
E-1 EXAMPLE GRAPHICAL SOLUTION FOR SINGLE-ROTOR GEARED GAS TURBINE HELICOPTER DESIGN HOVER CEILING 5000', DESIGN RADIUS OF ACTION 150 N.MILES, DESIGN PAYLOAD 3 TONS

The unique intersection of each pair of curves effectively equates the *required* and *available* R_F for a given hover ceiling, radius-of-action, payload, and disk loading, and locates the corresponding gross weight and R_F.

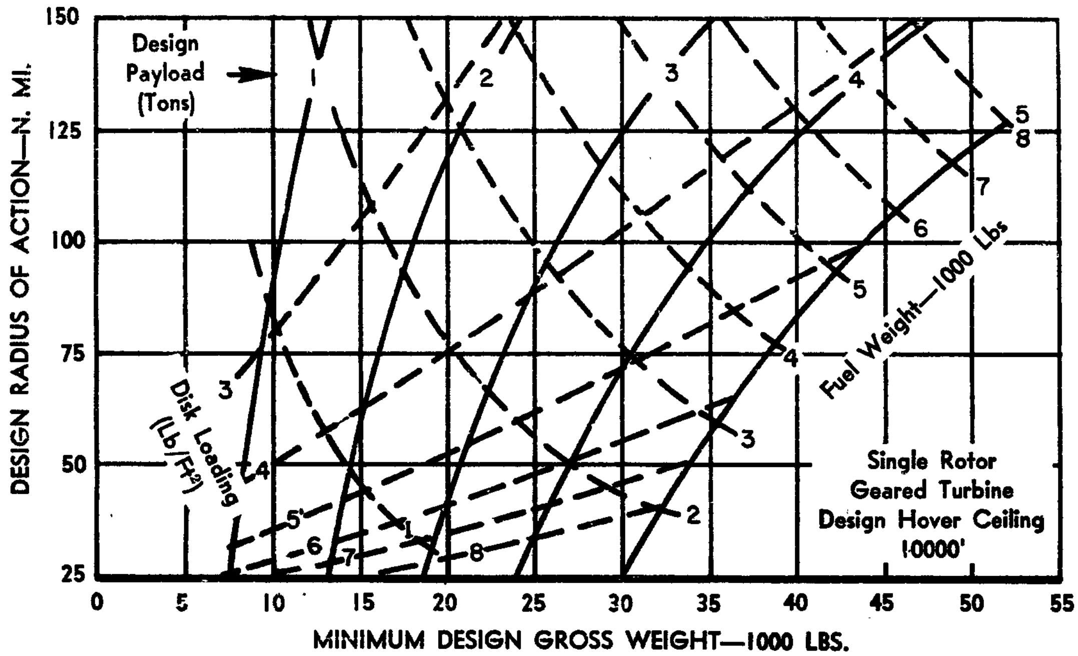
As shown in Figure E-1, the intersections for each disk loading are connected by one continuous *solution locus curve*, which indicates a *minimum gross weight* at some particular disk loading and R_F. It is from these locus curves that the design parameters have been established for all helicopters analyzed in this study.

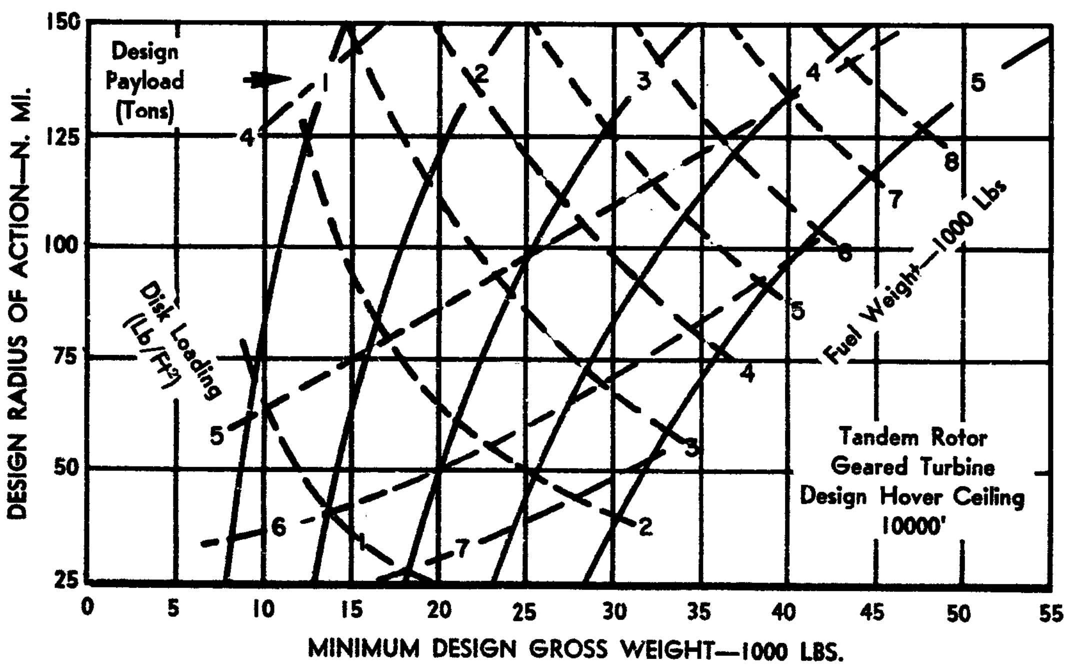
Appendix F
DESIGN CHARACTERISTICS CHARTS





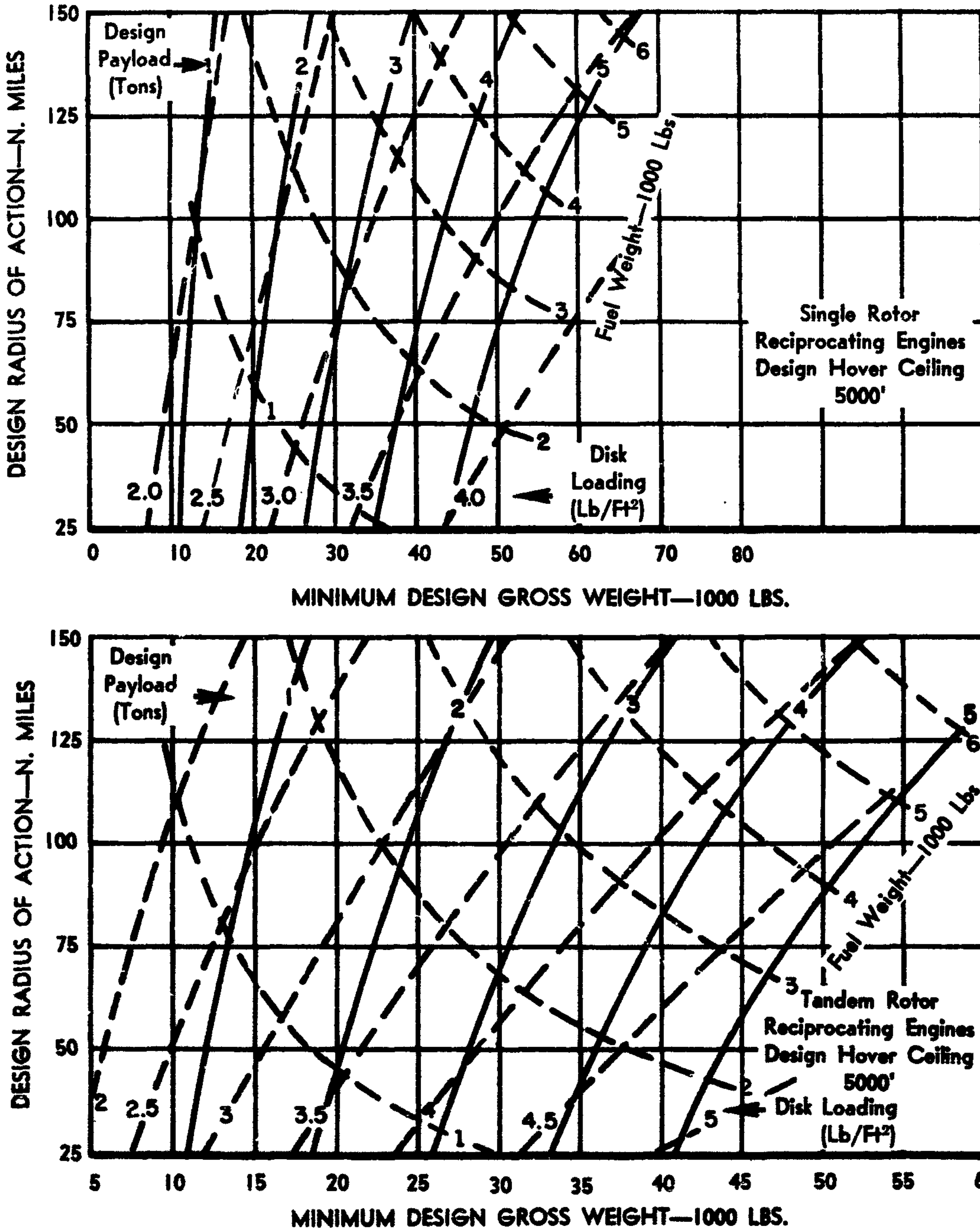
F-1 DESIGN CHARACTERISTICS CHARTS





F-2 DESIGN CHARACTERISTICS CHARTS

APPENDIX F—DESIGN CHARACTERISTICS CHARTS



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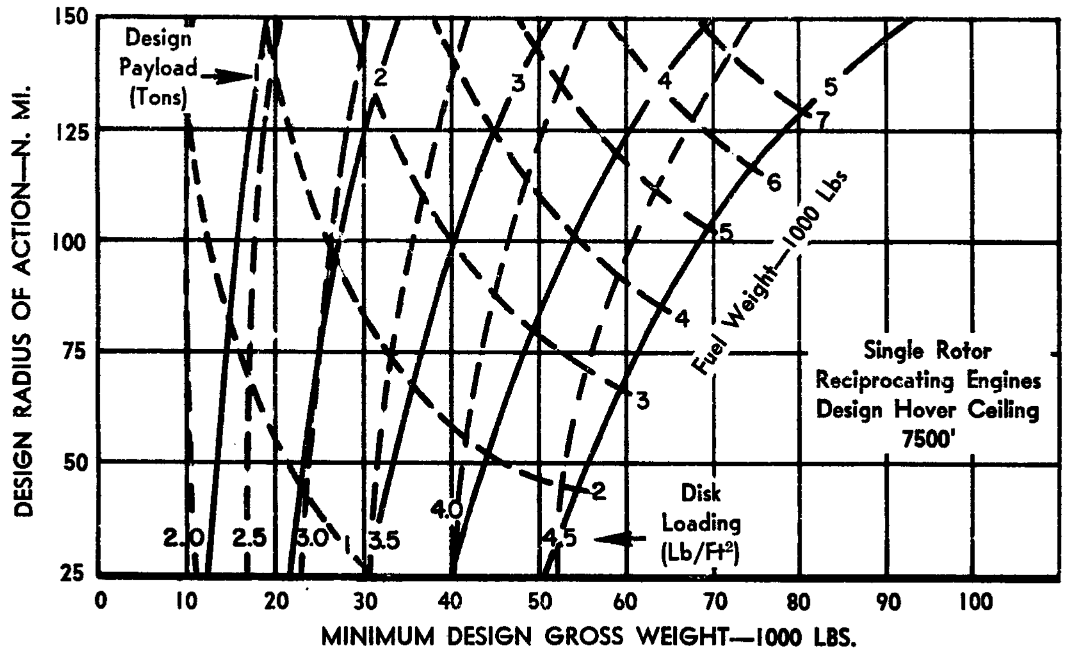
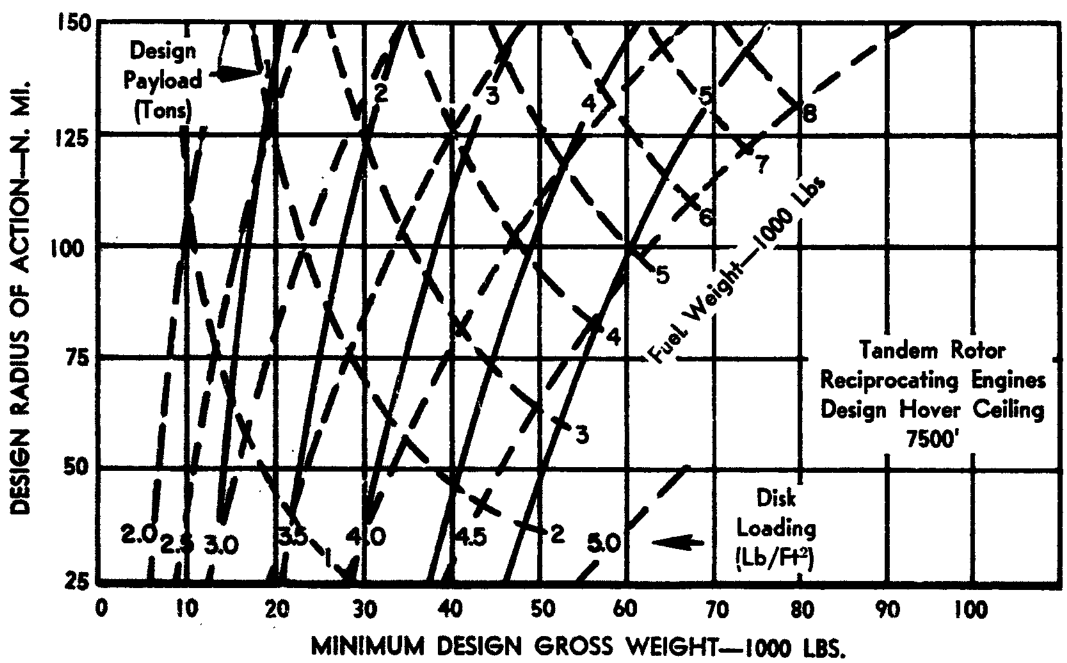
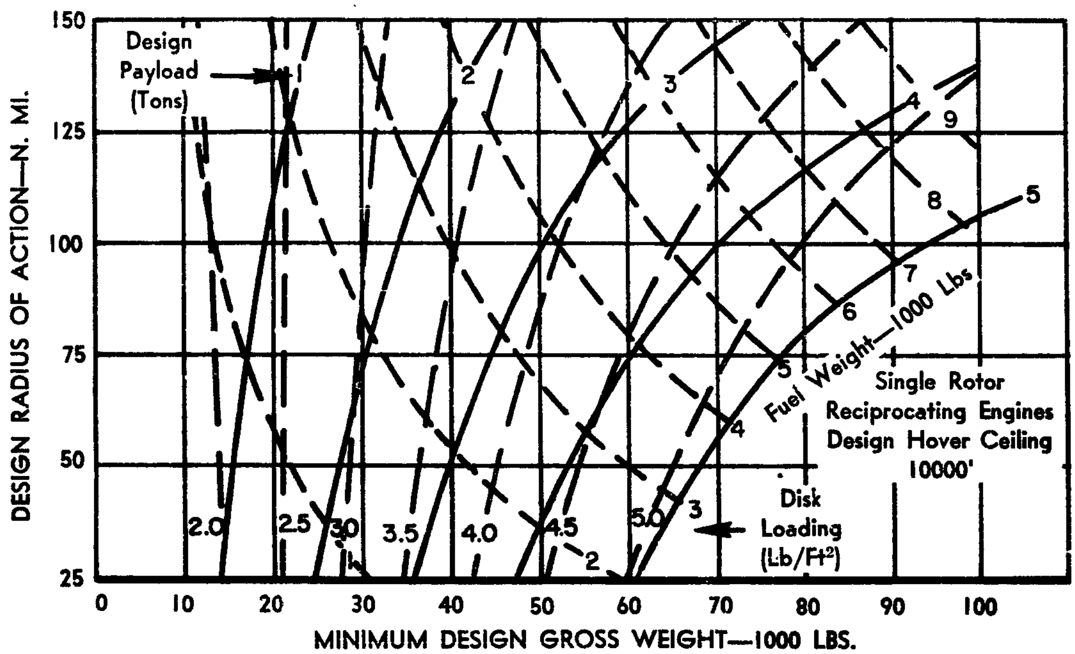
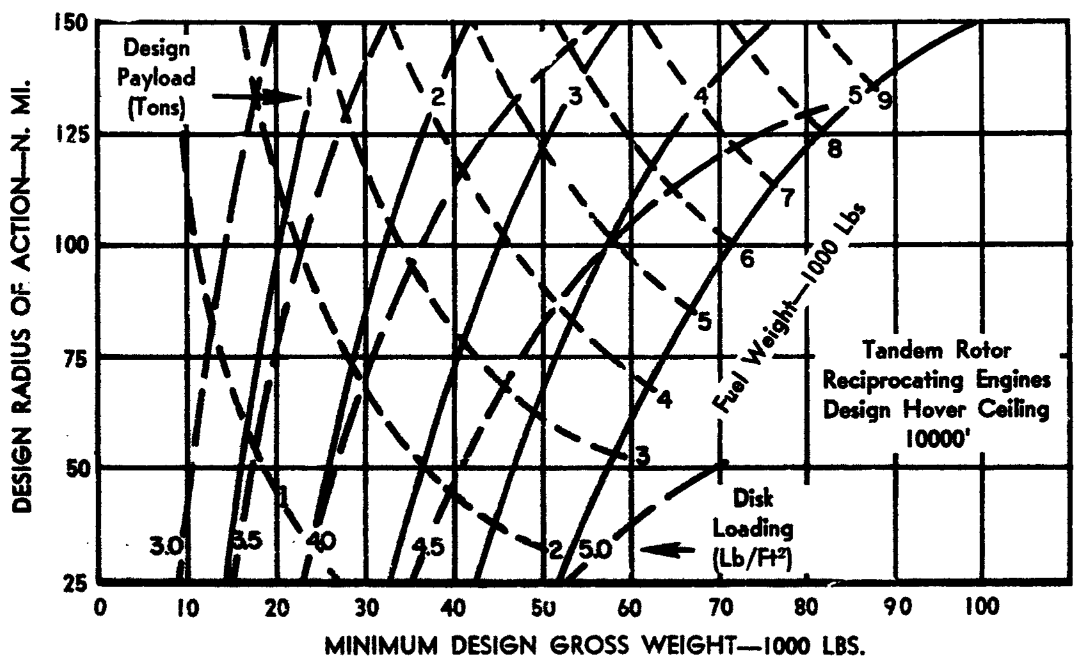




F-4 DESIGN CHARACTERISTICS CHARTS

APPENDIX F—DESIGN CHARACTERISTICS CHARTS





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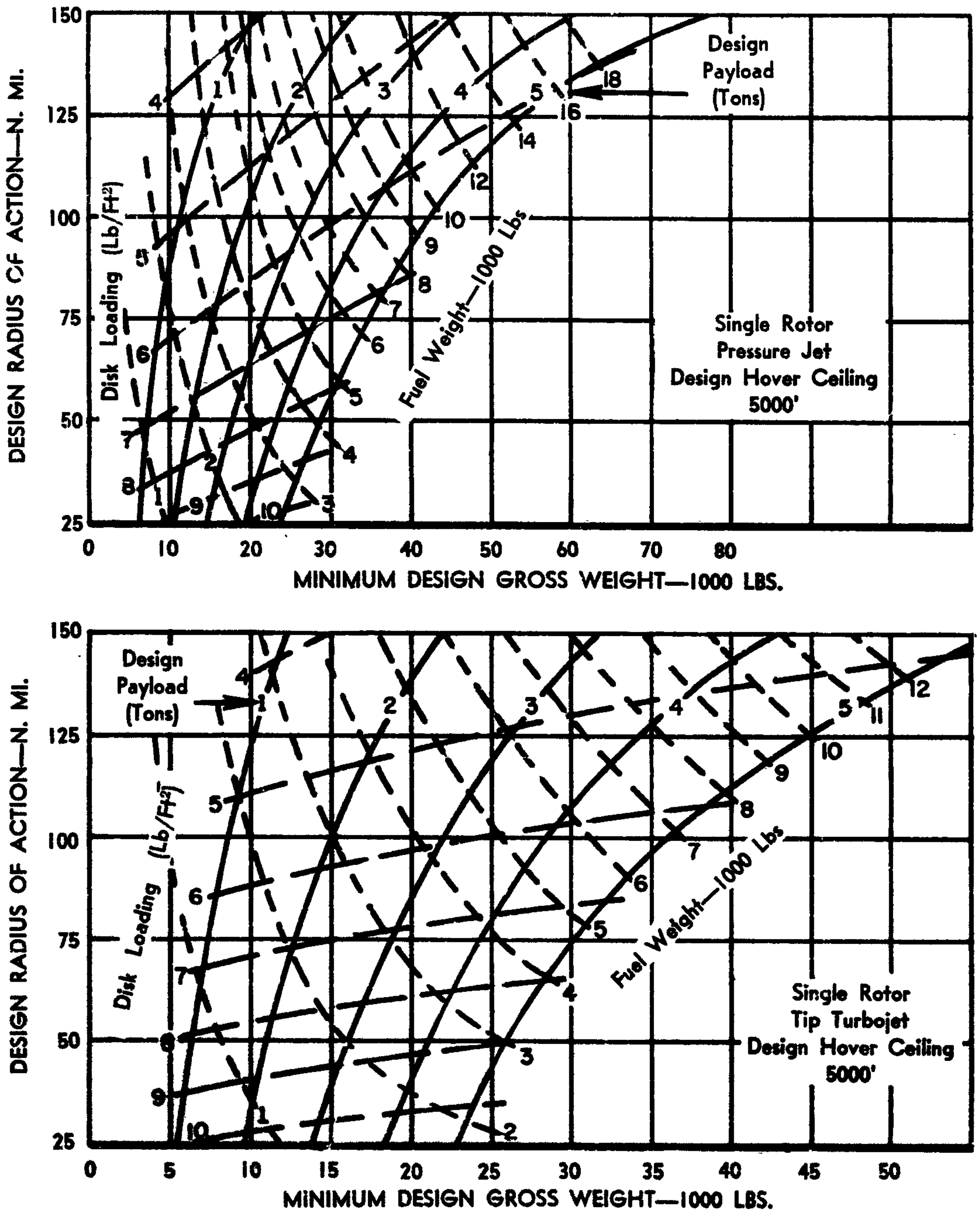
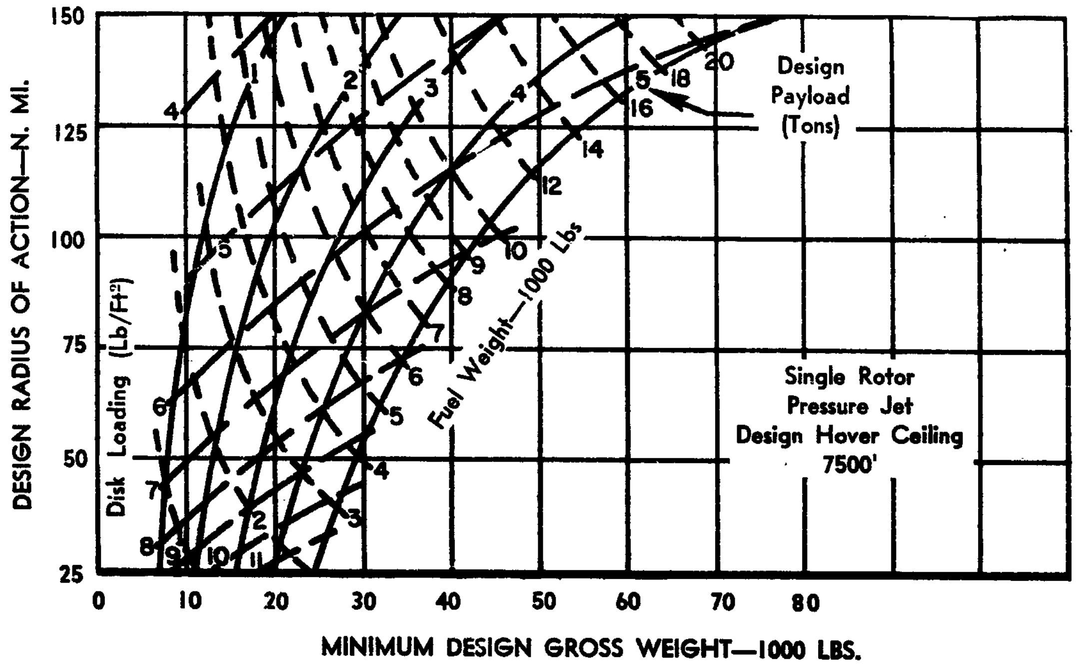
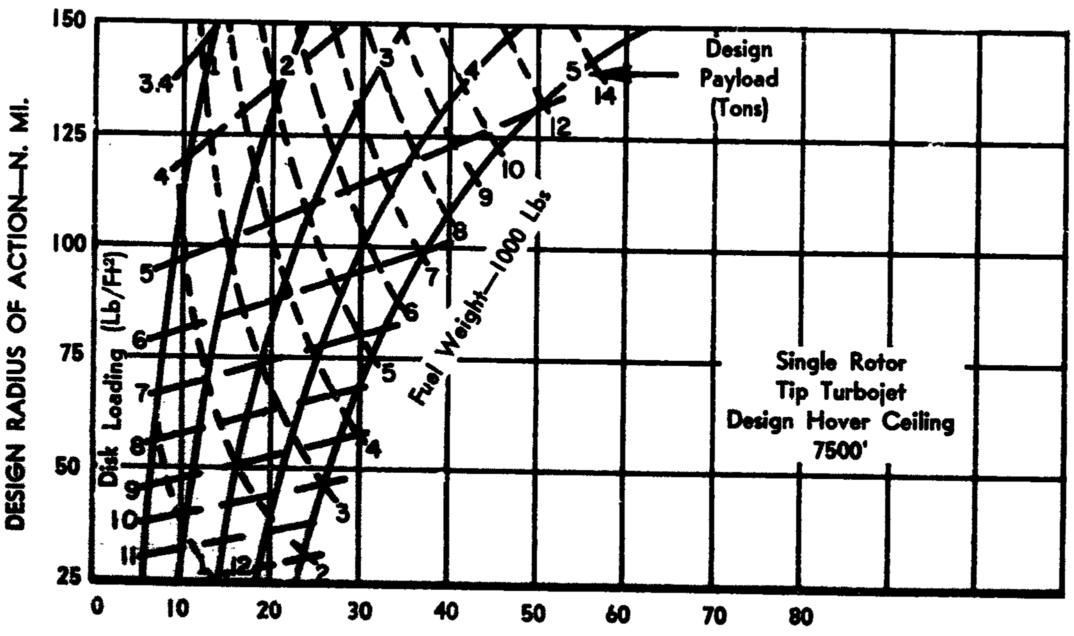


F-6 DESIGN CHARACTERISTICS CHARTS

APPENDIX F—DESIGN CHARACTERISTICS CHARTS





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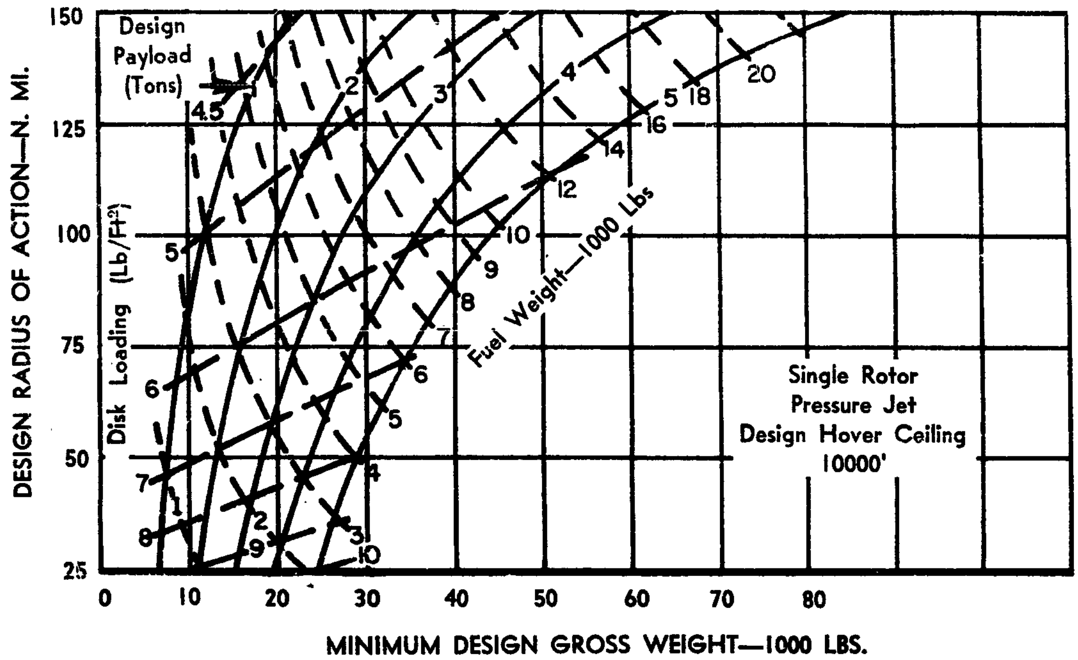
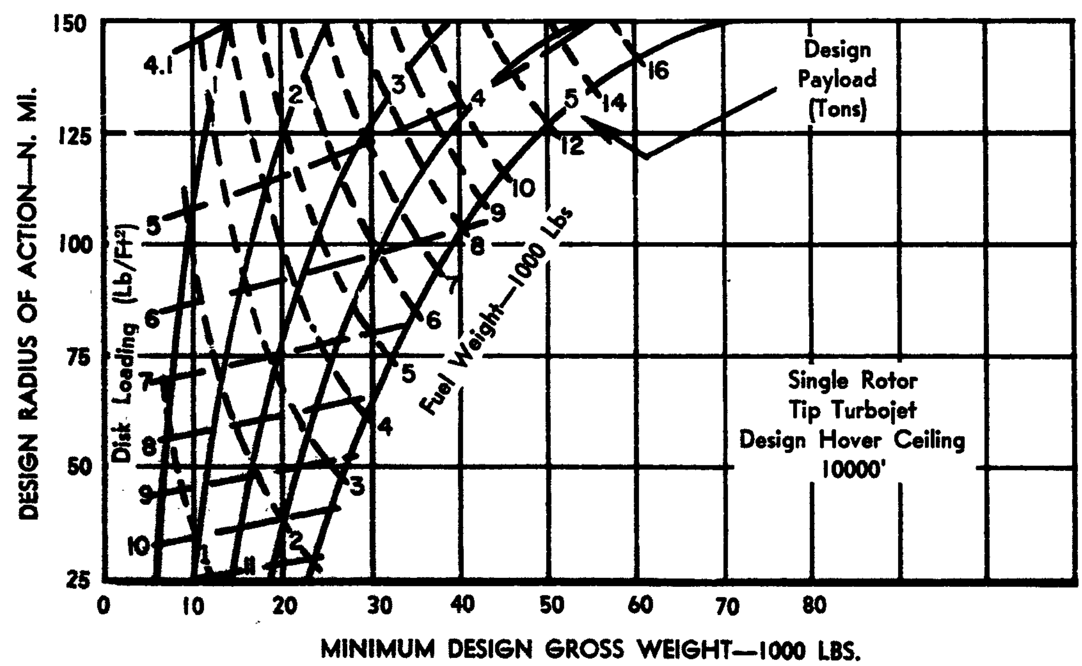
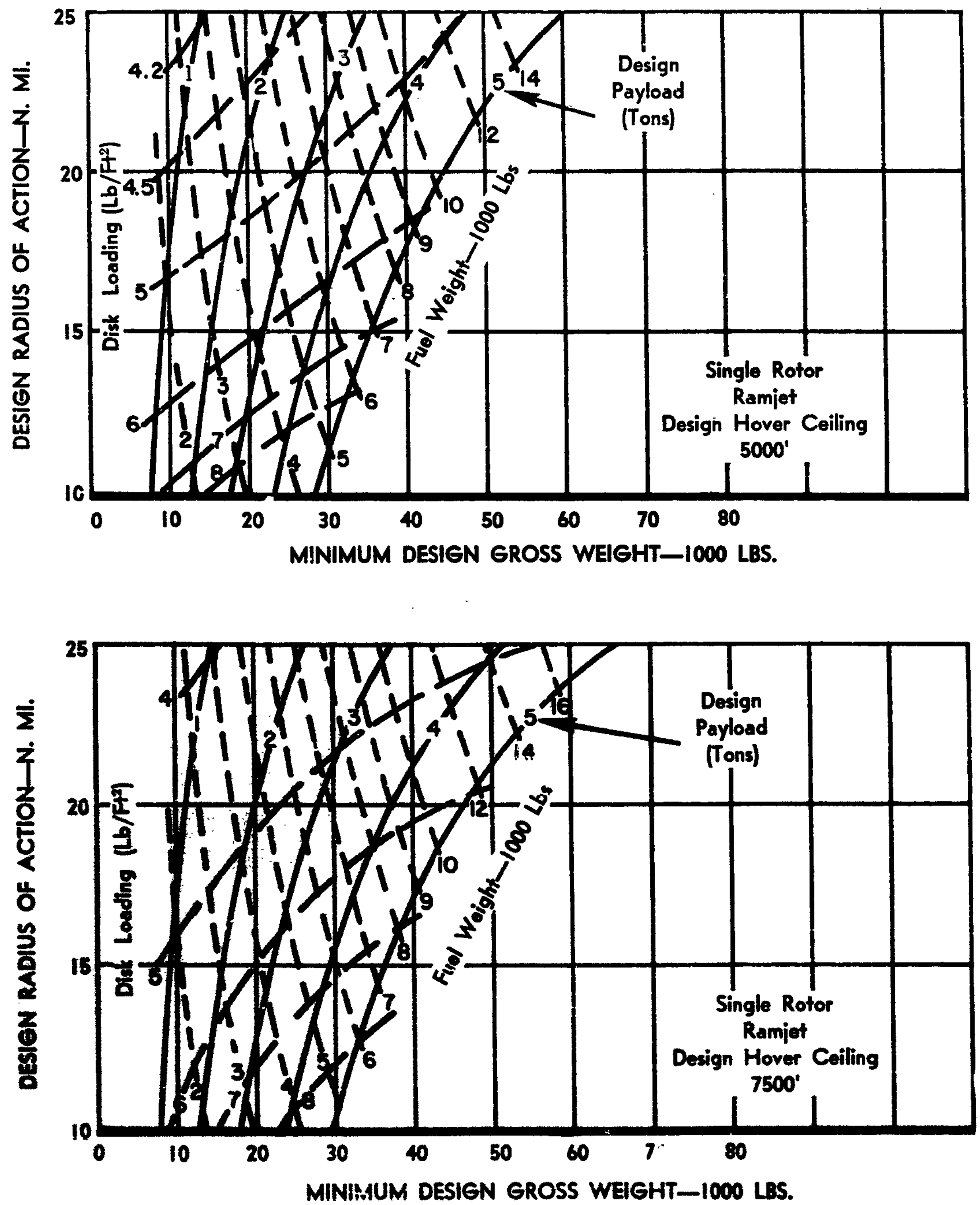


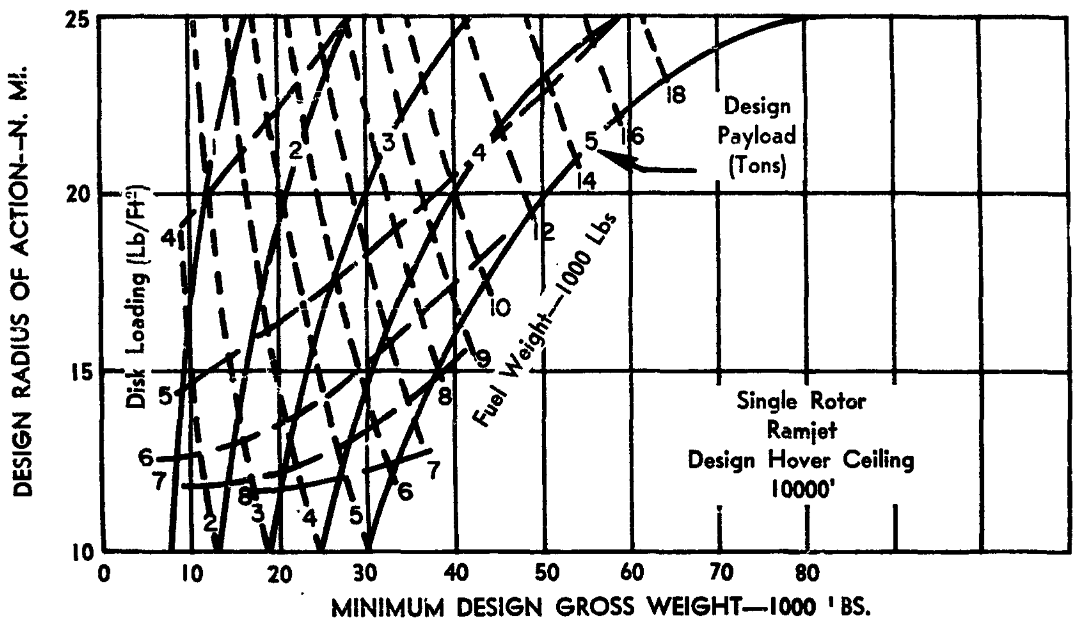


F-8 DESIGN CHARACTERISTICS CHARTS

APPENDIX F—DESIGN CHARACTERISTICS CHARTS



F-9 DESIGN CHARACTERISTICS CHARTS



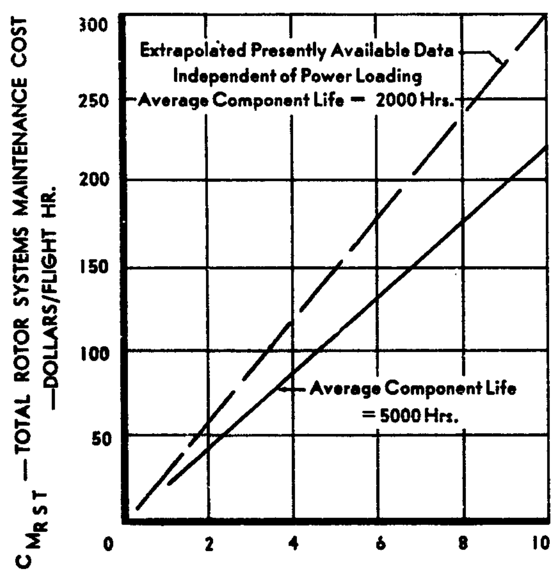
F-10 DESIGN CHARACTERISTICS CHARTS

Appendix G
HELICOPTER MAINTENANCE COST TRENDS

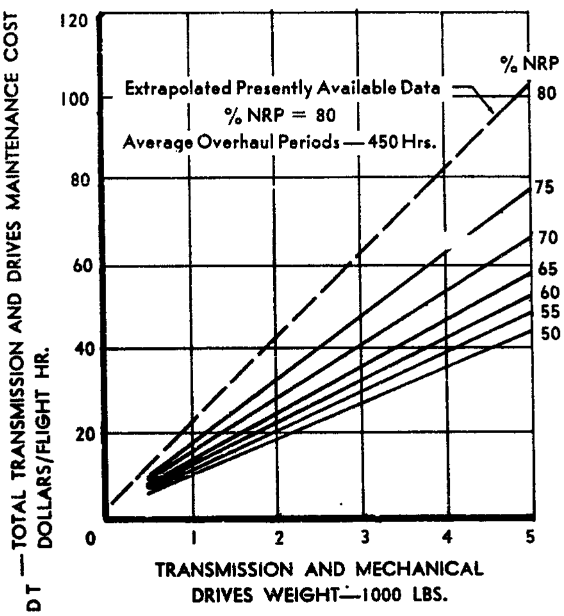
The maintenance cost data used in the evaluation of helicopter transport systems were based on available commercial operators' cost statistics. The flight hour cost trends with component group weight were found to be linear and are shown in Figures G-1 through G-7.

The maintenance cost charts for engines and transmissions and drives systems include the trends of flight hour maintenance cost vs. component group weight for various percentages of normal rated power required in cruise operation. The selection of the percent normal rated power for a particular design was based on charts of the type shown in Figure G-8. This chart covers single rotor types only, and is a general nomogram permitting the determination of percent NRP, for any combination of cruise speed, disk loading, equivalent parasite flat plate area per pound gross weight (A_π/W), and power loading. It may be noted that increasing either the drag area per pound (A_π/W) or the disk loading results in higher percent NRP in cruise at a given speed for a fixed power loading, and also, that lower power loadings reduce the percent NRP for a given A_π/W, disk loading, and cruise speed. Usually, the determination of cruise speed is based on maximum miles per pound of fuel, but the maximum speed limited by rotor compressibility and/or tip stall cannot be exceeded.

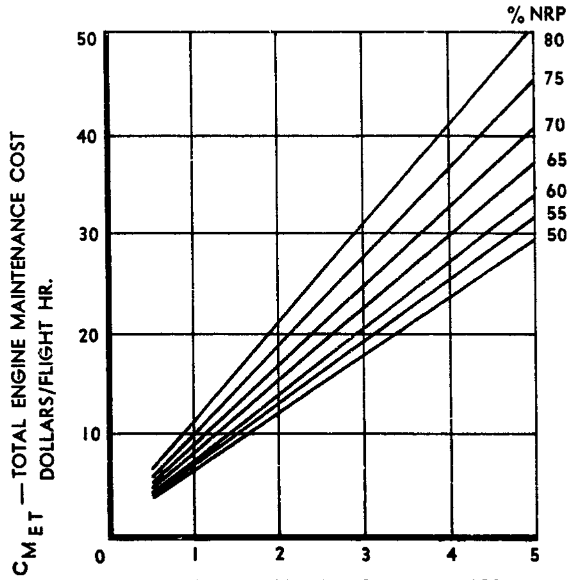
Figure G-1, for rotor systems, indicates the cost trends for both a 2000 hour and 5000 hour average rotor group fatigue life. The 5000 hour curve is corrected for the lower material costs attendant with greater component life. The maintenance costs used in the evaluations of this study were based on the 5000 hour curve.



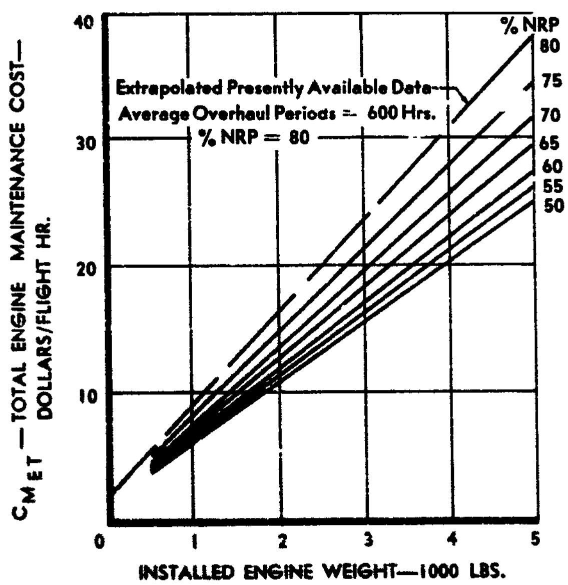
G-1 ROTOR SYSTEMS TOTAL MAINTENANCE COST



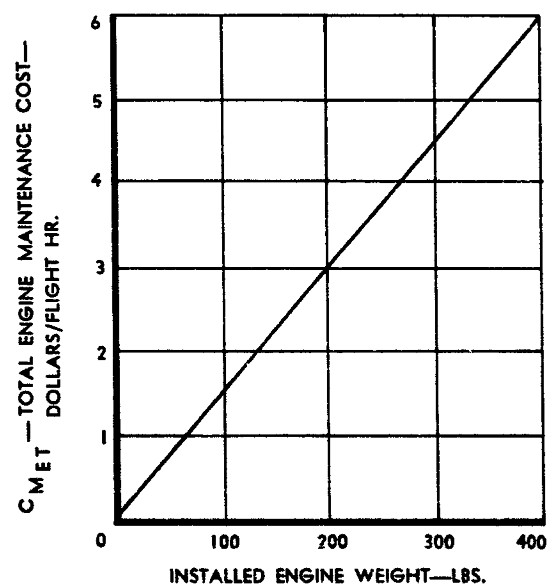
G-2 TRANSMISSIONS AND MECHANICAL DRIVES TOTAL MAINTENANCE COST



G-4 TURBINE ENGINES TOTAL MAINTENANCE COST

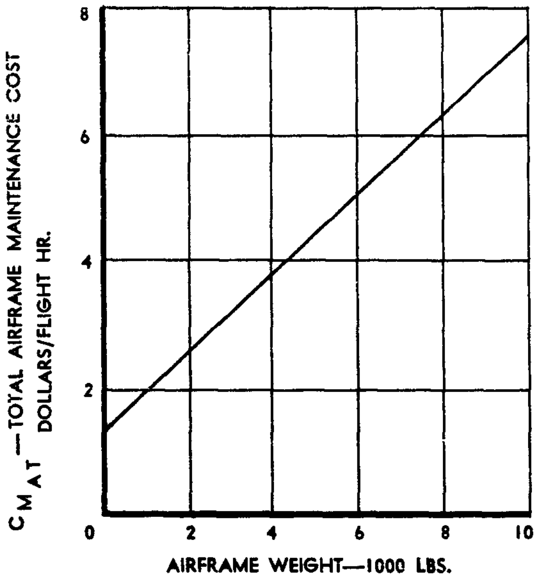


G-3 RECIPROCATING ENGINES TOTAL MAINTENANCE COST

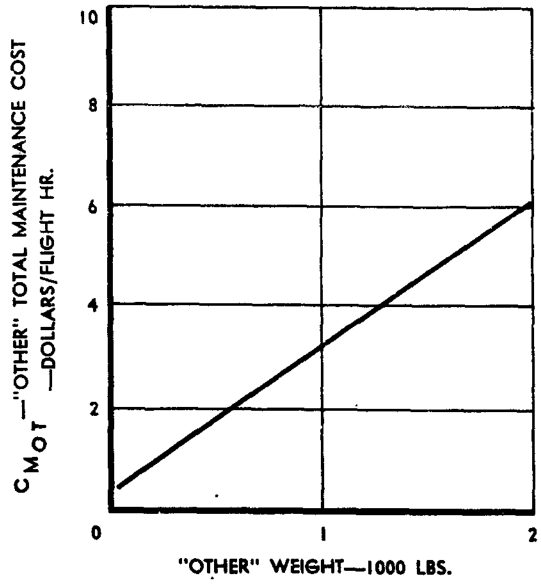


G-5 RAMJET ENGINES TOTAL MAINTENANCE COST

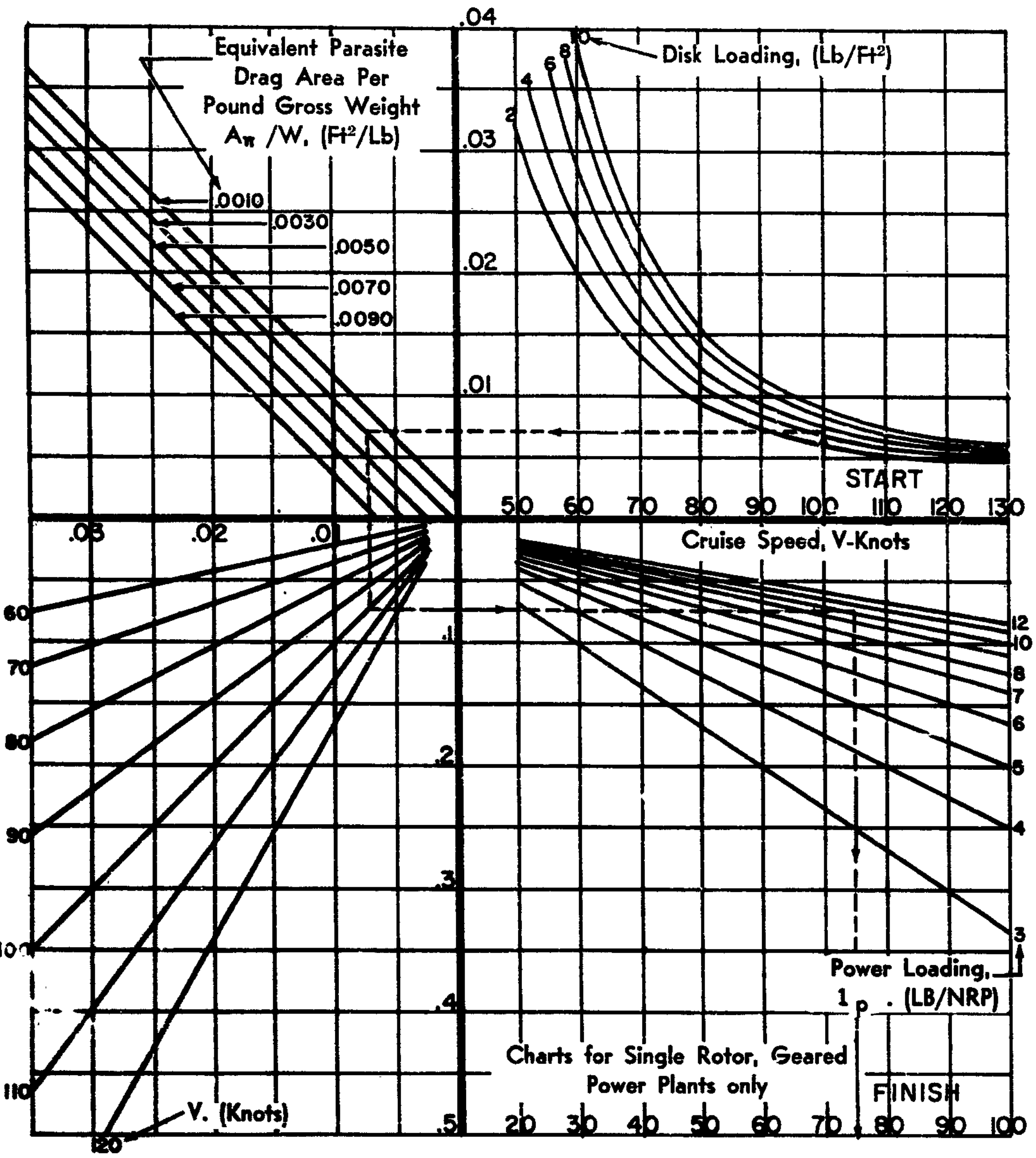
APPENDIX G- HELICOPTER MAINTENANCE COST TRENDS



G-6 AIRFRAME TOTAL MAINTENANCE COST



G-7 OTHER (RADIO AND INSTRUMENTS) TOTAL MAINTENANCE COST

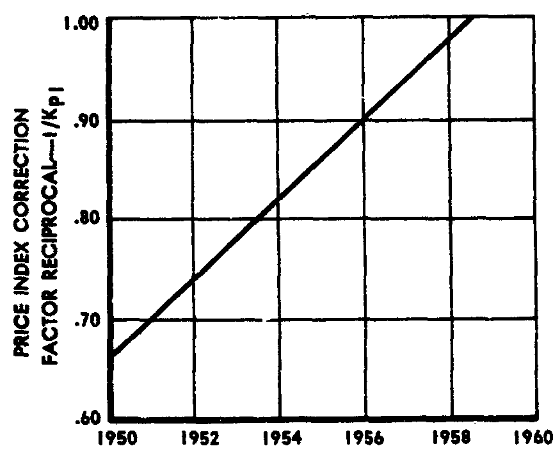


G-8 NOMOGRAPH SHOWING EFFECT OF POWER LOADING ON PERCENT NORMAL RATED POWER (VARYING WITH CRUISE SPEED, DISK LOADING, AND EQUIVALENT FLAT PLATE PARASITE DRAG AREA, PER LB., A_π /W)

Appendix H
PRICE INDEX CORRECTION

In the available data for the operational cost components, the dollar figures represented various economic levels depending on the year from which the statistical data was taken. In order to bring all costs to an equivalent level, a price index correction factor was employed. This factor was determined by examining the average hourly earnings of workers in the industries of aircraft and automotive manufacture[1] for the time period 1946 to 1954. The linear trend as shown in Figure H-1 was found and was extrapolated to the midpoint of the study. This allowed the adjustment of any cost data to mid-1958 by merely dividing the collected cost figures by the value of $(1/K_{PI})$ for the time period which the cost data represented.

[1] *Survey of Current Business.*



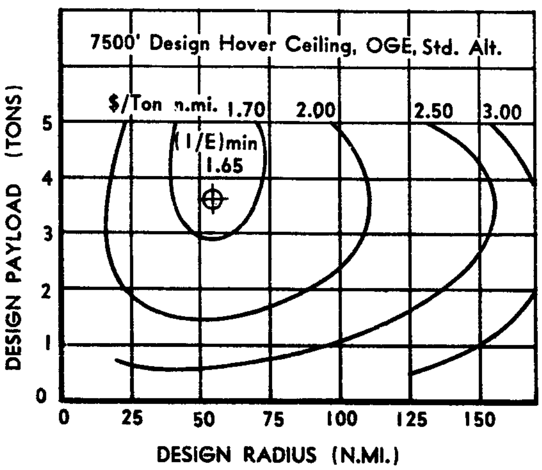
H-1 PRICE INDEX CORRECTION

Appendix I
SUPPLEMENTARY RESULTS

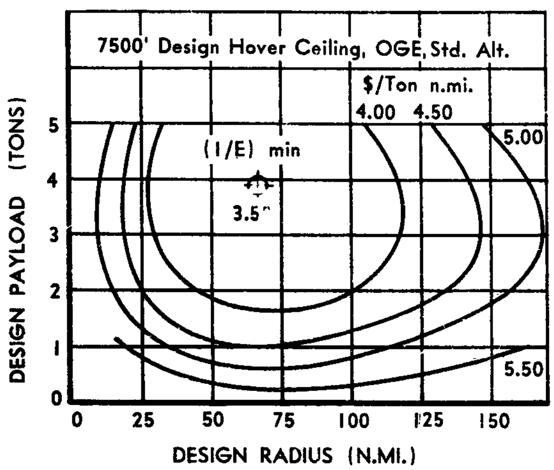
DESIGN PAYLOAD VS DESIGN RADIUS FOR VALUES
OF I/E—COST/TON N.MI.

- Single rotor helicopter
- Twin geared turbines

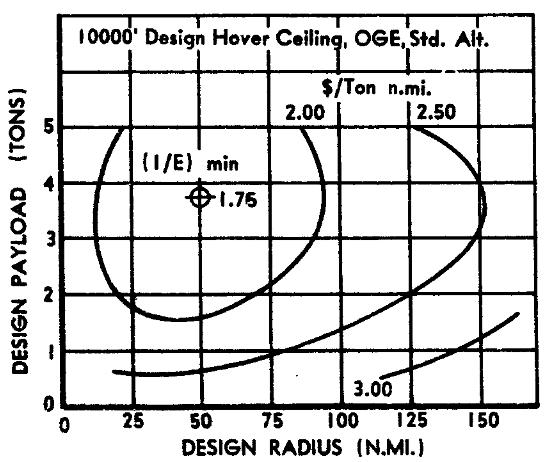
- Single rotor helicopter
- Twin reciprocating engines



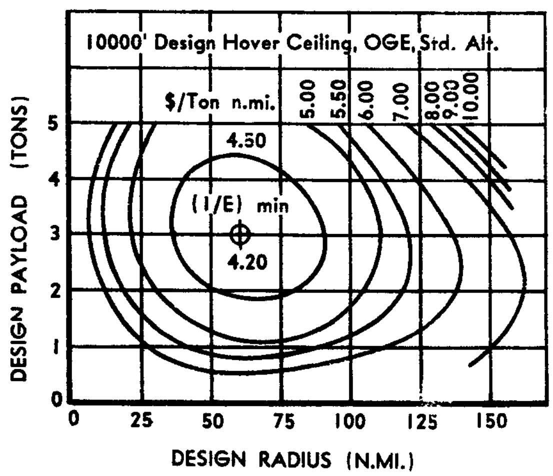
I-1



I-3



I-2

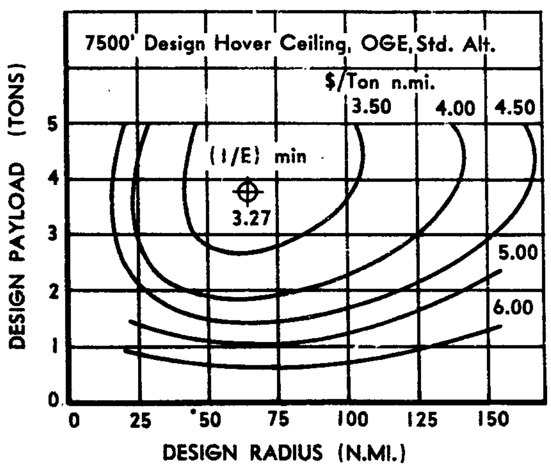


I-4

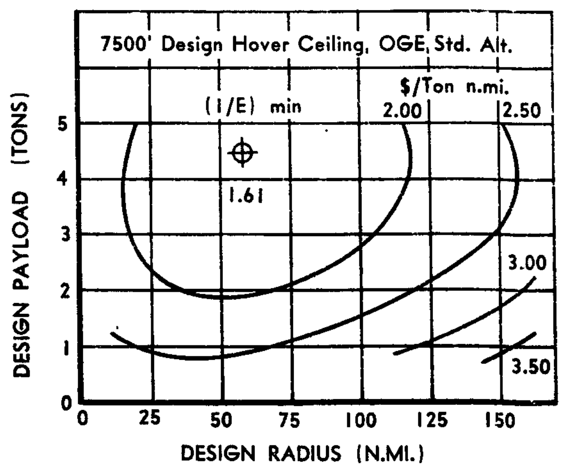
DESIGN PAYLOAD VS DESIGN RADIUS FOR VALUES
OF I/E—COST/TON N.MI.

- Tandem rotor helicopter
- Twin reciprocating engines

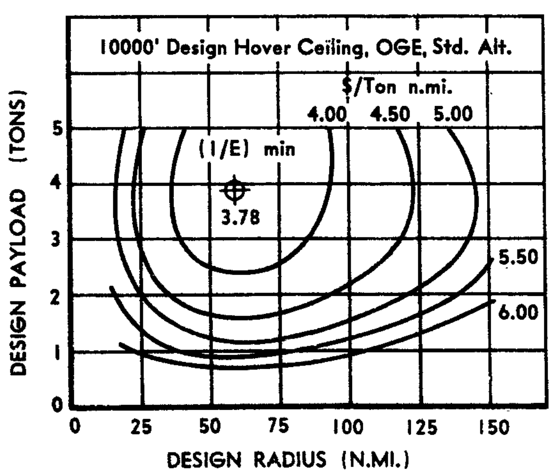
- Tandem rotor helicopters
- Twin geared turbines



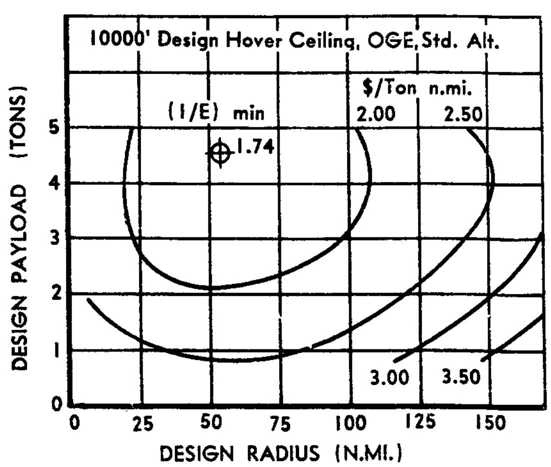
I-5



I-7



I-6



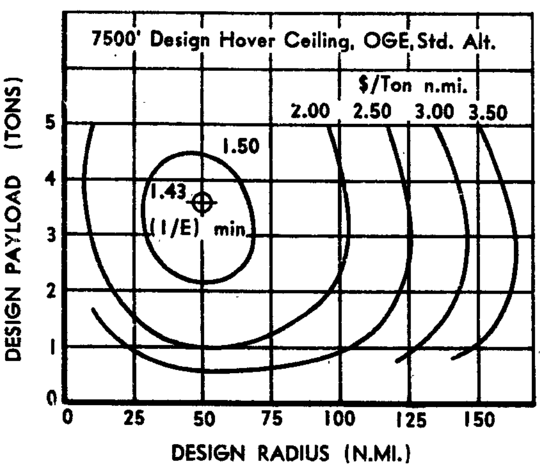
I-8

APPENDIX I—SUPPLEMENTARY RESULTS

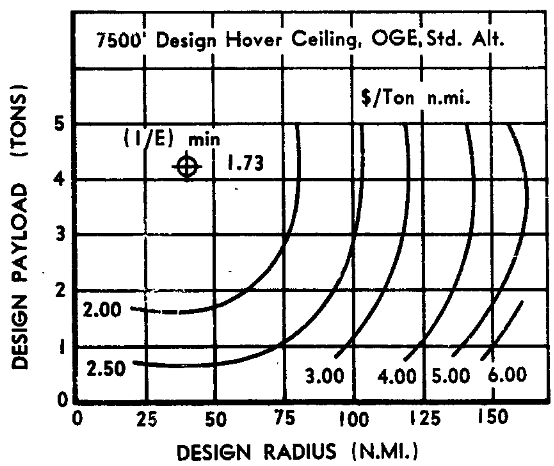
DESIGN PAYLOAD VS DESIGN RADIUS FOR VALUES OF I/E—COST/TON N.MI.

- Single rotor helicopter
- Tip mounted turbojet engines

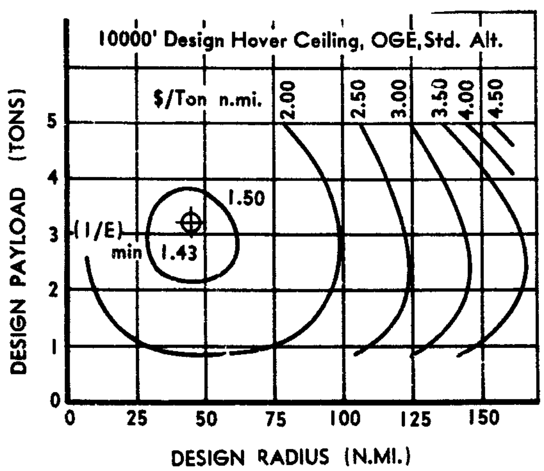
- Single rotor helicopter
- Pressure jet powerplant



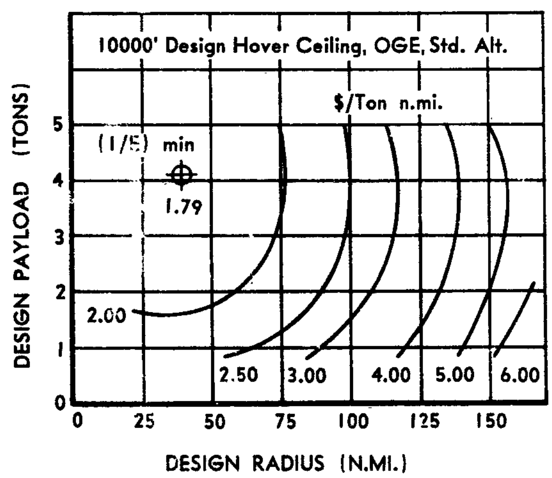
I-9



I-11



I-10

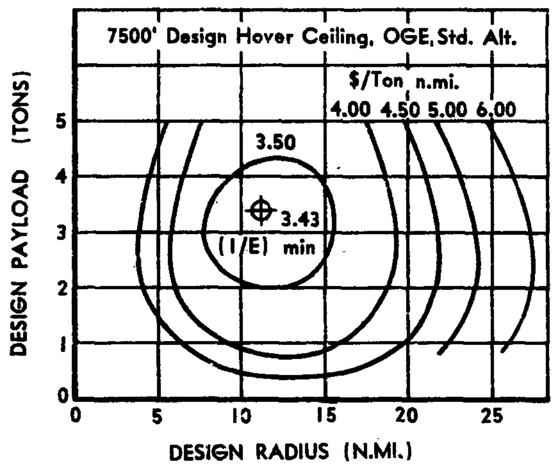


I-12

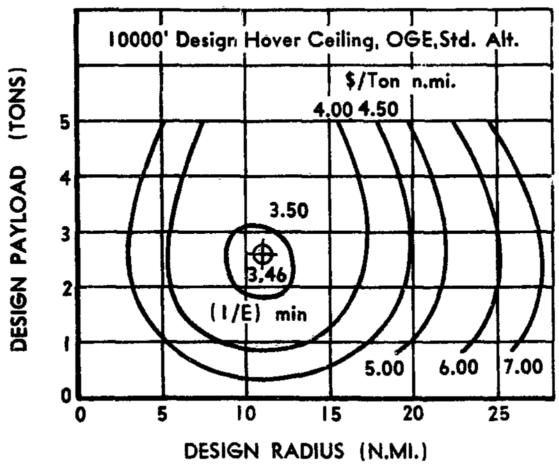
[redacted]

DESIGN PAYLOAD VS DESIGN RADIUS FOR VALUES
OF 1/E—COST/TON N.MI.

- Single rotor helicopter
- Tip mounted ramjet engines



I-13



I-14

[redacted]